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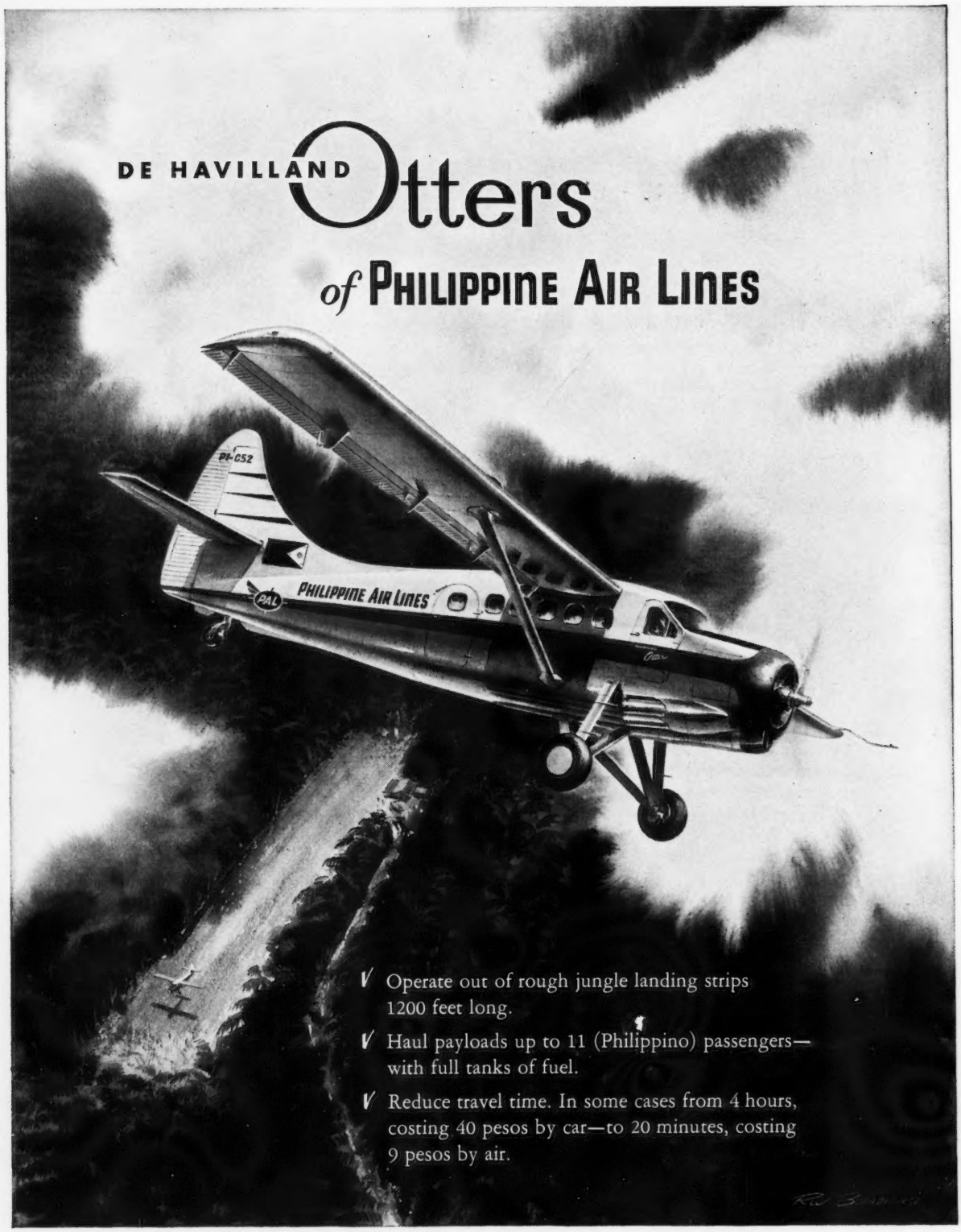
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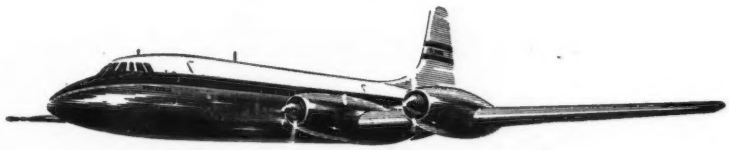

WHEN THE GAS TURBINE emerged about twelve years ago there was reason to suppose that its apparent simplicity would herald a profusion of different designs and so enable the new power form, with or without propellers, to be tailored to particular aircraft projects. Certainly in principle the engine is simple enough: a single shaft joins a compressor at the front to a turbine at the back. Between them is a continuously burning flame and the turbine collects energy from the heated and expanding gas to drive the compressor, leaving enough residual energy free to escape and propel the vehicle. But in the early days gas turbines often refused to work at all, and despite all encouragement they remained obstinately inert. Later, the gas was persuaded to provide enough power to turn the compressor, and gradually, after refinements, some positive thrust was produced. In due course, one pound of thrust could be generated for each pound of engine weight. This figure has steadily improved and now five pounds of thrust for every pound of engine weight can be achieved: indeed industry expects that later more than one pound of thrust will be obtained for every pound weight of the entire aircraft, so leading to fighters that can push themselves off the ground without forward run. Meanwhile improvements in fuel consumption have been made: until recently one pound of thrust required one and a quarter pounds of kerosene fuel, but now only one pound of fuel is needed. Naturally the penalty for these advances lay in increasing the maze of complexity. The compressor, for instance, instead of being forged from a single solid is now composed of hundreds of precision-formed blades—originally in aluminium, now in steel and titanium. The compressor of the 12,000 lb thrust Bristol Olympus embodies the further principle of duality—each half driven through concentric shafts at different speeds by separate turbines with only a gas connection between them.

Time has shown that the task of evolving a big gas turbine demands fifteen to twenty times the effort that went into the most complicated piston engines. A high proportion of these additional engineering and design man-hours has been spent in unremitting research toward a better understanding of how to handle enormous volumes of air. Instead of using fifteen parts of air to one of fuel as in a piston engine, the gas turbine uses eighty-five parts of air. And naturally the science of guiding air into the engine through the intake with minimum losses is as important as reducing duct losses and turbulence when it is inside the engine. Apart from the aerodynamics, the science of turbine metallurgy has also been in a state of constant change. Early in the career of the gas turbine, aluminium alloys had to be replaced by various chrome and cobalt steels and nickel alloys. Some of these have scarcely had an opportunity to prove themselves before they, in turn, have been overtaken by the properties of titanium, whose immensely important alloys are midway between aluminium and steel in weight and approaching steel in strength.

The result is that new types of turbine engines are now taking some five to seven years to develop, and the gap is widening between the capabilities of 7-10,000 lb engines in production and those with three times the power which are now feasible. Discoveries follow each other in quick succession and there is no hint anywhere that this characteristic pace in turbine technology is slackening. The Engine Division of the Bristol Aeroplane Company is balancing the rapidly expanding frontiers of scientific knowledge with the cycle of production that follows long afterwards in its wake.

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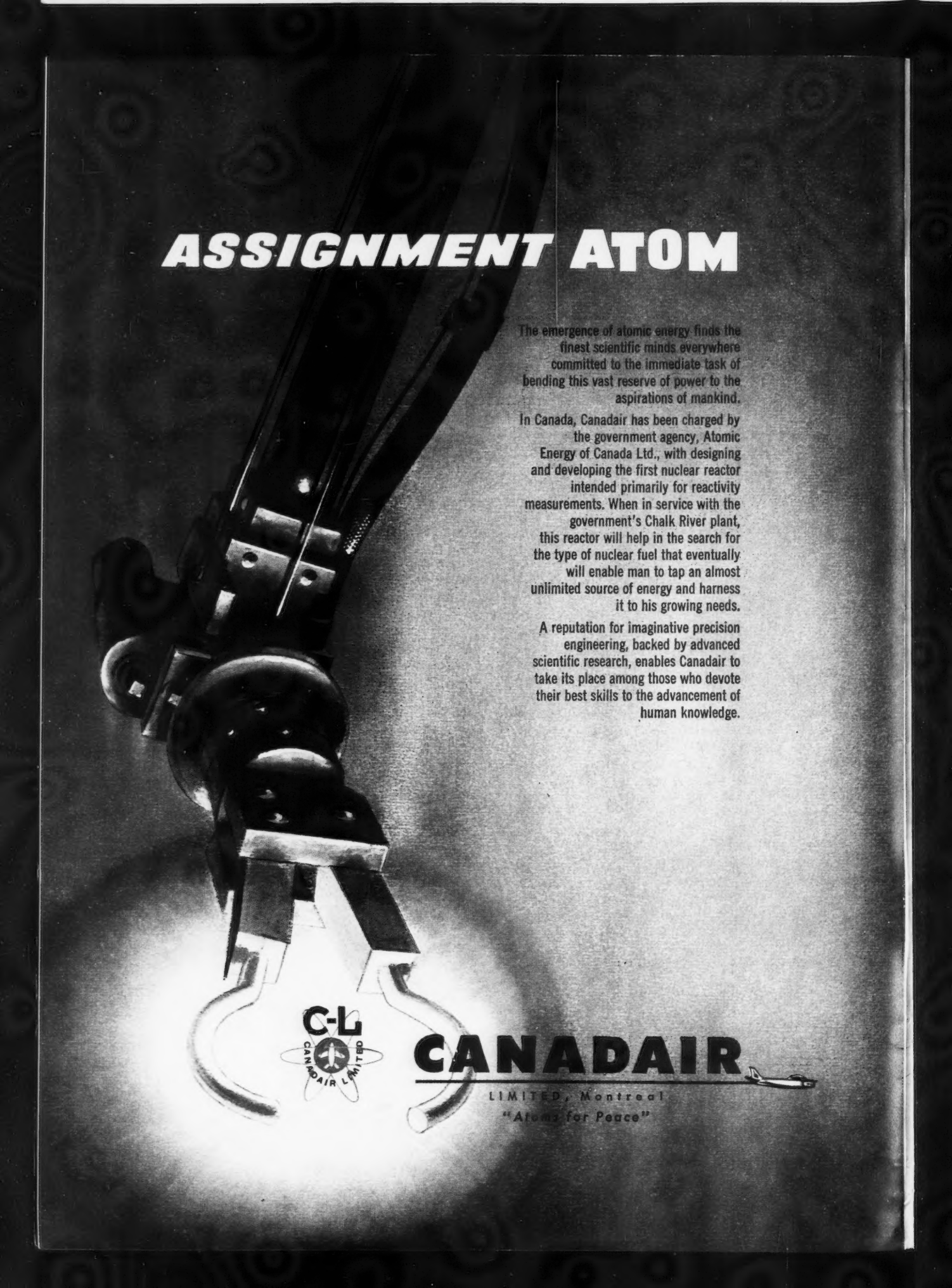
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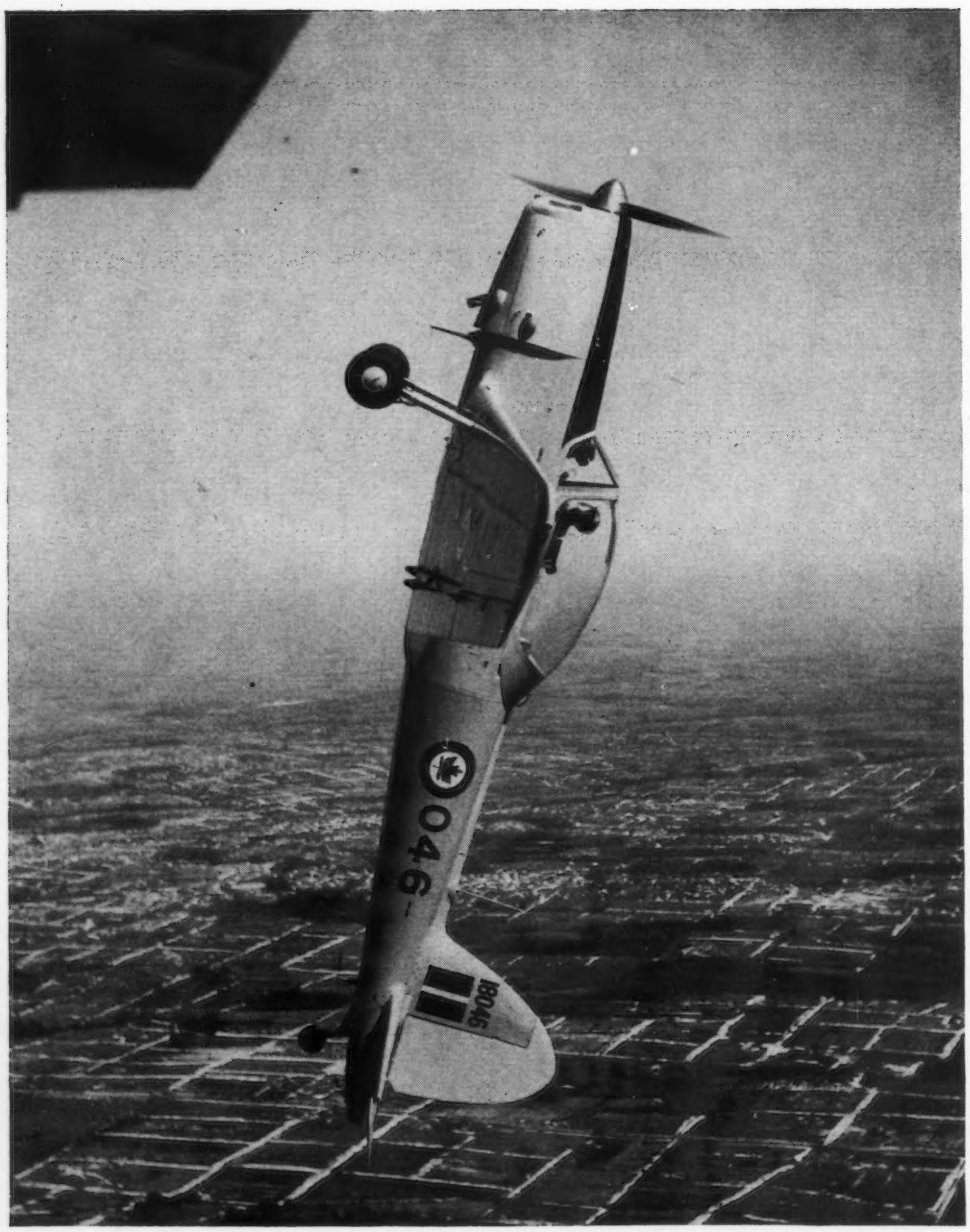

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EDITORIAL

HUMAN FACTORS IN AIRCRAFT DESIGN

EVERYONE seems to believe in the important part that studies on human factors should play in aircraft design. Yet here in Canada we do not see the kind of partnership that should result from such a belief. Partnership is necessary, rather than consultation and advice, if human abilities and limitations are to be reflected adequately in the result.

The day has passed when reliance can be placed solely on common sense in this field. It is not enough to design the machine and then bring in the consultants; too much time and money will already have been spent for the ready acceptance of important changes. Specialists in human factors too often find themselves in the position of patchers, trying to make the best of a badly designed man-machine system.

Many glaring examples in the past indicate that human limitations and advantages have not been fully recognized. "Pilot error" is still blamed for lives lost, when the real fault may lie in poor cockpit design, and "inadequate maintenance" takes the blame for what is often bad aircraft design for maintenance.

Aircraft are changing rapidly and radically, but man changes very slowly. Consequently, the low percentage of aircrew candidates who reach operational combat standards is hardly surprising. Because aircraft are increasingly complex, and because the chances and penalties of error are much greater, it is urgent that we work out a more effective man-machine combination than before.

We cannot expect to overcome such difficulties completely. But we can do much to meet and even anticipate the problems of fitting aircraft to human capacities by a clear recognition of them in the organizations that design and build, and operate and maintain aircraft.

Perhaps the present situation results simply from the obstacles to communication between specialists. In this country, the aeronautical engineers and those working on the human side – the physiologist, the work methods specialist, the psychologist, the medical man – have not usually worked as teams. The engineer and the aircraft operator have called in the human factors expert only when they saw the need for help. Although such consultations may be valuable, they can never take the place of partnerships. As a first approach, many difficulties of communication could be avoided if a competent engineer were to be made directly responsible for all human factors elements in the design. He would have the advantage of being able to communicate easily with his colleagues, and he would soon find how and where he could obtain the specialist support and advice he would have to have.

What is needed now, in Canada, is a working partnership of aeronautical and human factors specialists. It is not too late to recognize this need in both the customers' and suppliers' organizations. A complex man-machine unit, such as an aircraft with its aircrew, deserves design and operating organizations that provide this partnership.

M. G. WHILLANS, M.D.
Assistant Chief Scientist and Director of Personnel Research, Defence Research Board.

VENTURI BLEED-OFF

During a project of functional tests of a Lancaster aircraft at Central Experimental and Proving Establishment, Rockcliffe, it was found that a large cut-out on the port side of the nose section caused severe vibrations of the adjacent fuselage skin and buffeting of the tail plane. The oscillations of the elevators could be clearly felt at the controls. The cut-out, for camera installation purposes, was about four feet long and thirty inches wide. The sides were parallel to the aircraft centre line and the rear face was a vertical plane at right angles to the plane of symmetry.

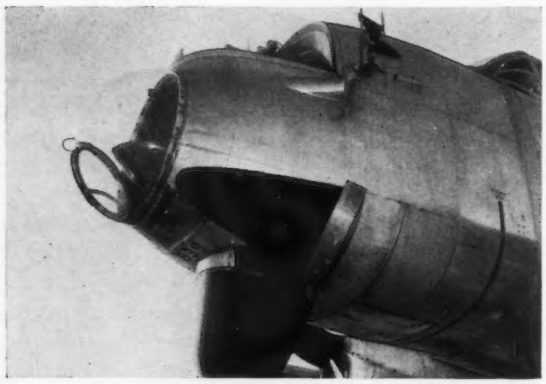
The disturbances were caused by pressure waves building up at the rear face of the cut-out and discharged at the side of the fuselage at a frequency of the order of five cycles per second.

There are several ways of correcting this fault and the following were considered.

(a) Deflectors or some form of vortex generator in the front of the cut-out.

(b) Bleed-off holes at the rear face of the cut-out.

(c) Bleed-off by venturi at the side of the fuselage at the rear of the cut-out.



General Arrangement

The first method, (a), had to be abandoned because of interference with the camera line of sight. The second, (b), proved to be impracticable because the ducting would interfere with the camera brackets and other installations. The economic aspect also made this solution unattractive.

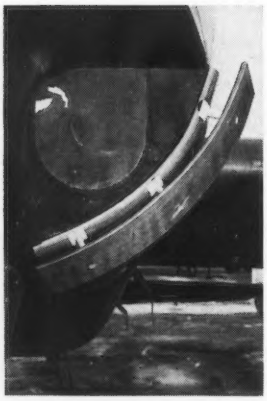
The third method, (c), was chosen for the following reasons.

(i) It would give adequate suction to prevent pressure waves building up in the rear of the cut-out.

(ii) It was simple and inexpensive to produce.

(iii) It could easily be fitted on the aircraft.

Rear view



Front view

The venturi was made up of two airfoil sections of laminated wood, the upper cambers facing each other, and spaced two inches apart. The inside airfoil section was made to fit the curvature of the fuselage. The outside section was inverted, mounted at an angle of attack of approximately eighteen degrees, and placed two inches forward of the inside section. The spacers were made of streamlined steel tubing, with bearing plates mounted on each end. The assembly was then securely mounted on the side of the aircraft in the plane of the rear face cut-out. The installation mounted on the aircraft is shown most clearly in the picture giving the rear view.

The aircraft was tested at various speeds, up to 300 knots in dive, and the flight test reports showed that all vibration and practically all buffeting had been eliminated.

No tests were carried out in icing conditions. It is expected, however, that icing will occur and the problem has been discussed to some extent. In order to prevent ice formation in the venturi, the sections could be made of metal with heating elements installed, but the sections, having composite curvature, would then be complicated to produce. The inconvenience of icing was not considered serious enough to justify the use of metal construction. Even if ice builds up on the airfoil sections, the vibrations due to this particular cut-out would scarcely be discernible from those caused by the many other iced-up protuberances.

This project showed that the venturi principle or diffuser can be used successfully to prevent local build-up of pressure waves on an aircraft due to large cut-outs. However, if it should be desirable to use the venturi bleed-off on high speed aircraft, further tests would have to be carried out.

H. Aass
C.E.P.E. Rockcliffe

AIRCREW RECOVERY FROM AIRBORNE MISHAPS†

by S/L E. D. Harper*

Royal Canadian Air Force

OBJECT

THE object of this paper is to outline the problems of recovering the aircrew who are involved in emergencies while airborne. This is not a result of any morbid ideas on flight safety; after all, if test pilots were upset at the thought of these emergencies, they would not be in the jobs they are. From a practical point of view though, it is only reasonable to want to stay alive with a minimum of injury and besides there is a certain amount of economy involved in rescuing the high priced help.

HISTORICAL REVIEW

This problem first cropped up some time ago. Ancient Greek mythology mentions a couple of people who suffered as a result of flying accidents—one encountered an overheat problem and his wings broke off and the other, Bellerophon, is the first man on record to be shot down by a guided missile, when Jupiter damaged Pegasus with his gadfly. In both cases the pilots suffered from lack of adequate safety equipment. Then, of course, da Vinci was concerned with the problems of flying and parachutes. The need for a positive means of airborne rescue was seriously felt in 1914-1918 when flying became more frequent. The need was not so serious as it is today however. I have reports of people without parachutes involved in various escapades and surviving quite handily. One I heard of had the engine fall out of his aircraft, while the aircraft collapsed in a heap from about 1,000 ft. He walked away with nothing worse than an evil smell—he had landed on a manure heap.

Parachutes first came into serious use about 1919, although they were used for sport before that. In fact, the R.C.A.F. standard Irvin parachutes were designed then. In those days, with slow aircraft, using this parachute was not much of a problem; the user merely had to climb out of his seat, walk to where he could safely clear the aircraft and then jump. Barnstormers used to make quite a show of it and, in Russia, balloon jumping was considered quite a sport to while away an afternoon.

In the last war, the situation began to get more difficult. Some of the latest piston type aircraft, such as the Spitfire and Mustang, were capable of reaching speeds which made it virtually impossible for the pilot or crew to get out in event of an emergency. This obviously was not recognized as a serious problem, or at least it was not treated seriously, and we find aircraft, such as the Vampire, Venom and some other jet aircraft, flying even at this late date with no means of getting out of them except to climb out. It was not until some extremely capable and valuable men were killed that much was done to solve the problem of escape from high speed aircraft. And even now, we are in the situation where emergencies can occur in flight from which the pilot has little or no hope of escaping without serious injury. The fact is that this problem has been pretty well ignored all along and developments in the field of crew safety have not kept up with the advances made in the capabilities and hazards of aircraft.

PRESENT AIRCRAFT AND EQUIPMENT

In the high altitude, high speed field in Canada, we now have in use the Fury, Mustang, Vampire, Meteor, Banshee, T-33, Sabre and CF-100 in military flying, and Mosquito and Lightning in civil aviation.

The piston engine aircraft, although not high speed in the modern sense of the phrase, are included in the list because they are all capable of reaching speeds and altitudes at which the physiological problems are so great as to make abandoning the aircraft safely almost impossible with their present equipment. They, the Vampire and Meteor, offer the least in the form of emergency provisions for the crew. At the other end of the scale, with the most advanced escape equipment, is the CF-100.

THE PROBLEMS OF EMERGENCY ESCAPE

At this time, it would be well to mention the problems involved in emergency escape from this type of aircraft. These come in various flight conditions,

(a) High altitude
(b) Low altitude
(c) High speed
(d) Low speed

and can be considered as either "deliberate", where the pilot recognizes an impossible situation well in advance and reasons himself into a nylon letdown, such as might occur in impossible weather conditions with no fuel—a good example is the recent 6 Hunters in Norfolk—or

†This paper was presented on the 2nd March 1956 to a meeting of service and civilian test pilots at C.E.P.E. Rockcliffe. It was submitted to the Canadian Aeronautical Journal on the 17th April 1956.

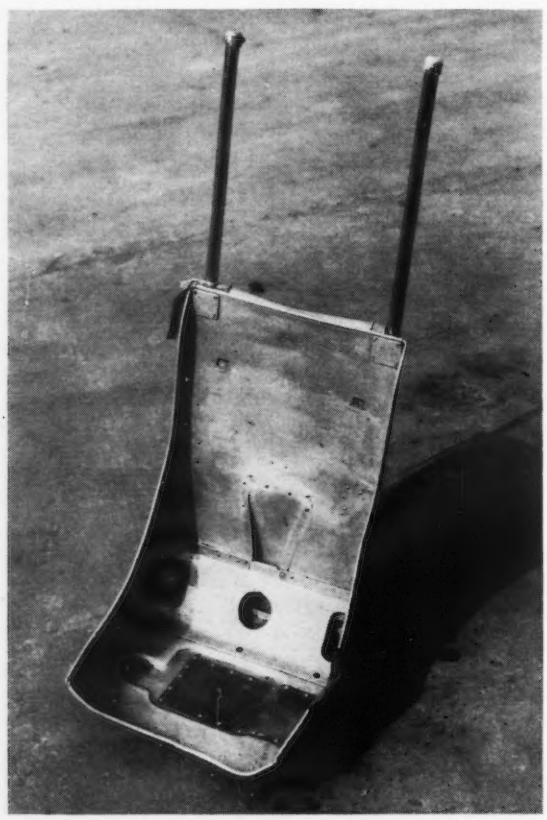
*Deputy Chief Test Pilot, C.E.P.E., Rockcliffe.

an "urgent" escape where conditions could be practically anything. Furthermore, the actual escape can be considered in 3 phases—the departure from the aircraft, the trip down and the arrival on the ground.

Any equipment that will cope with the urgent escapes will also cope with the deliberate. Therefore, this paper will discuss only the urgent conditions.

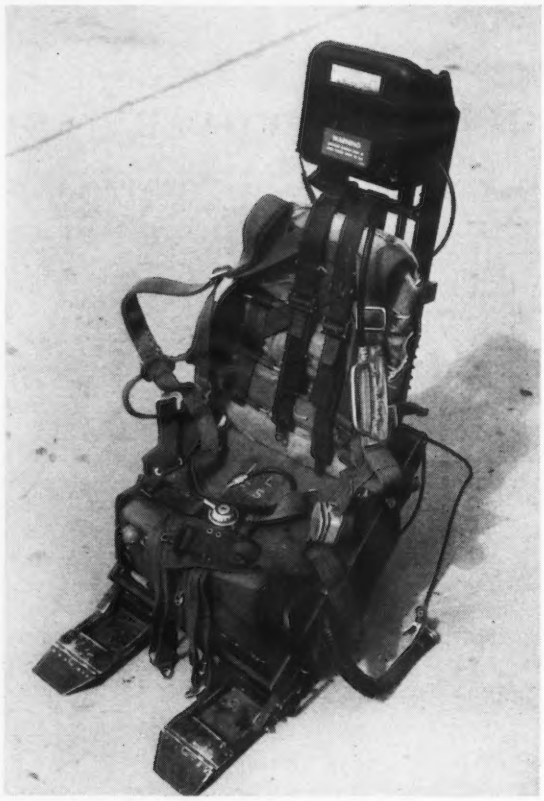
Low Altitude—Low Speed

First, the low altitude, low speed case. On the face of it, this appears relatively simple. It is complicated, however, by the type of aircraft being considered—things are still pretty fast and emergencies have a peculiar way of compounding themselves. Some examples of this are engine failure in the Sabre, accompanied or followed by electrical failure and difficult circumstances; engine seizure in the Sabre or similar power controlled aircraft; fire in the air; or low level engine failure over poor country. The fact is that there have been a large number of people killed in low altitude, low speed circumstances, who could have been saved by adequate equipment. The first phase has been pretty well solved for all conditions—you pull a handle and, bang, you are out of the aircraft. However, in the low altitude case, what happens nowadays leads to the third phase being rather unfortunate—the arrival on the ground has been somewhat sudden.



Harvard bucket seat

This photo illustrates the standard bucket seat as fitted to Harvard, Mustang, Mosquito and Vampire aircraft. A seat-type parachute is required and in emergencies the pilot must undo the seat harness and then attempt to climb out of the cockpit and jump free of the aircraft.



T-33 Ejection seat complete with flex-back parachute and emergency pack

A standard 24 ft flex-back parachute is illustrated which necessitates an additional set of harness for securing the pilot to the seat. This type of seat is manually released by the pilot after ejection, but is currently undergoing trials for conversion to automatic release and parachute deployment.

The obvious requirement is very quick reactions to get the parachute open in time. This requirement is being met fairly well in the CF-100 by means of an automatic parachute operating system. The other aircraft do not have it yet. To cover every eventuality in low level escape, a timing of between 1 and 2 seconds will permit safe ejection from ground level if the aircraft is travelling between 120 kts and 250 kts.

Low Altitude—High Speed

At speeds over 250 kts, however, the parachute opening shocks on the man using a time delay of 1 to 2 seconds are prohibitive. Because of this, the timing has been set to 5 seconds on the CF-100. This covers the low level conditions on our best present day equipment—at very high speeds, you get a fairly severe but acceptable opening shock—you are protected down to approximately 200 to 300 ft above the ground.

High Altitude—High Speed

Because of the mechanics of the automatic gadgets available, certain aspects of low speed and high speed escape are similar. A barostatic device opens your parachute at a predetermined height—R.C.A.F. sets 12,000 ft. This permits you to reach a reasonable parachute opening speed.

EJECTION SEATS

The various ejection seats in use all meet the requirements of the first phase—swift positive departure from the aircraft. They give between 10g and 20g and you reach an ejection speed of approximately 60 ft/sec. The Martin-Baker seat, as in the CF-100, provides automatic parachute opening which copes with all speeds the aircraft can reach between 200 ft and as high as the aircraft can go. The T-33 and Sabre seats will soon have similar devices. The parachute, once it is safely opened, takes care of the arrival on the ground. However, another major problem still exists on our present day aircraft. That is the problem of what happens to the man between the time he clears the aircraft and the time he is safely hanging on the parachute. At average and low speeds, this is not a serious problem. At high speeds though, there are major problems which have yet to be solved. Air blast effects, violent contortions and high acceleration loads cause very serious injuries. On the T-33 and Sabre, this is most serious. The Martin-Baker system, however, does give the maximum protection available at the present time. The seat provides support and protection and a stabilization parachute prevents the wild tumbling which occurs on the other seats.

Martin-Baker automatic seat as fitted into CF-100 aircraft

The illustration shows the standard 24 ft seat style parachute which necessitates a double-harness, one for the chute and one for the seat. The seat stabilizing parachute is housed in the seat superstructure. A more recent Martin-Baker seat employs single harness, leg restraints instead of foot rests, an integral firing system and a duplex stabilizing parachute designed for high speed, low altitude ejections.

DEVELOPMENT IN EQUIPMENT

There are various modifications to all this equipment which will make it more useful. The T-33 and Sabre seats are getting automatic devices to improve them. This will not, however, protect the pilot against injury from air blast and tumbling or flailing around. Another mod for them will be a stabilization parachute. The Martin-Baker Mk 3 seat has further mods consisting of leg restraint devices and a modified firing system. This, at the present time, as I said before, affords the most protection now available.

None of these offer protection against air blast and acceleration effects for high speed bail-out. Escape capsules have been mentioned, in various magazines, but there is still the problem of high acceleration forces to be solved and these capsules are only on paper.

Another point that has not been mentioned is that of surviving the cold and lack of oxygen from high altitude bail-outs. Protective clothing comes in here, but that is another subject. Portable oxygen bottles solve the problem of lack of oxygen but this solution, however, is still relatively primitive. The idea of having to strap a bottle onto your parachute and pull a round knob if you have time, or think about it soon enough, illustrates exceedingly well how very little planning and forethought have gone into the problems of survival in emergencies. If the parachute can be automatic, why not make the oxygen automatic?

CONCLUSION

The problems of emergency survival have always, in the past, been recognized by a hind-sight system of development and, as a result, the development of protection of crews still lags behind the development of new aircraft.

I do not want to appear morbid about these things and I feel we do have some reasonable equipment now, if the limitations are recognized. I would like to see more effort made to get aircrew protection planning into the aircraft planning stage on a realistic basis.

SOME AERODYNAMIC STUDIES IN THE C.A.R.D.E. AEROBALLISTICS RANGE

by Dr. G. V. Bull*

Canadian Armament Research and Development Establishment†

SUMMARY

The aeroballistics range technique as employed at the Canadian Armament Research and Development Establishment is described and some experimental results presented. The experimental results are shown to illustrate the type and quality of data obtained. The technique itself consists in launching models from a smooth-bored gun by means of a sabot carrier and measuring their flight trajectory by means of yaw cards surveyed into position. By fitting the equations of rigid body motion to the experimental trajectories, the aerodynamic derivatives are obtained. The experimentally determined variation with Mach number of aileron power, damping in roll, pitching moment, normal force, center of pressure, damping in pitch and drag is presented for a cruciform wing-body combination. Illustrative results are also included on other types of configurations.

INTRODUCTION

In order to study the performance, stability and control characteristics of bodies moving through dense media, the Newtonian equations of motion may be applied. These equations involve on one side force and moment terms and, on the other side, terms of mass and inertia. By judicious choice of reference axes, the mass and inertia terms may be evaluated readily. The force and moment expressions, however, present a much more difficult problem, even when the most favourable axes-system is chosen. The problem here is two-fold: firstly, it is necessary to identify the independent variables and combinations of independent variables contributing to forces and moments and secondly, it is necessary to find the functional relationships between these independent variables and the total force and moment. Solving the first problem is a mixture of art, science and experience, with the latter predominating. Once the independent variables and combinations thereof are identified, the force and moment functions may be approximated to any desired order of accuracy (and complexity) by use of a Taylor expansion in these variables and their time derivatives. In general aeronautical nomenclature, the force and moment derivatives with respect to the independent variables and their derivatives are termed the "stability derivatives". The differential equations of motion may be written and solved once these expansions have been determined. No analytic solutions to these general, non-linear equations exist and to solve them it is necessary either to consider restricted cases or to use automatic computing devices.

†Received 20th March, 1956. A revised version of a paper read at the C.A.I. Annual General Meeting, in Toronto, on the 20th May 1955.

*Section Head, Aerodynamics.

Figure 1
Interior view of the Aeroballistics Range

The stability derivatives may be determined experimentally from either static (such as wind tunnel) or dynamic (free-flight) tests. The principles of these two experimental techniques are antithetic. In wind tunnel testing the stability derivatives are measured statically and substituted into the general equations of motion to predict free-flight response to changes in the independent variables. In free-flight testing the dynamic trajectory resulting from some combination of independent variables is measured and by fitting the general differential equations of motion to the experimentally

determined flight path, the stability derivatives may be deduced. It is apparent then that a knowledge of the trajectory equations is fundamental to both methods. The free-flight method has the advantage, perhaps, that while erring in the estimation of aerodynamic coefficients, it still satisfies the resultant motion.

Normally free-flight testing of full scale objects, such as aeroplanes and missiles, is extremely costly both in time and money and wind tunnel tests are used to enable accurate prediction of the performance to be expected in free-flight tests. In this scheme of things, free-flight testing may be considered as acting in a confirmatory role. However, by the use of free-flight model ranges it is possible in many cases to perform dynamic, free-flight model tests on a cost and time basis comparable to that of static wind tunnel tests.

These ranges have evolved from the early proof testing techniques of the ballisticians[a] and are generally classified under the titles of Ballistics Range, Aerodynamics Range and Aero-ballistics Range. There are approximately half a dozen such ranges known to be in operation outside Canada and one in this country. Since they differ significantly in instrumentation and technique this paper will deal only with the Canadian range.

DESCRIPTION OF THE C.A.R.D.E. AEROBALLISTICS RANGE

The interior of the C.A.R.D.E. aeroballistics range with the original frame holders in place is shown in Figure 1. The range proper is a building of concrete construction approximately 20 ft × 20 ft in cross-section and 750 ft long. One end of the building is closed except for the inlet hole through which the models enter. This closed end is protected by a blast wall and armour plate, the latter to stop carrier fragments. The other end of the range is open, permitting the models to continue in flight into a large sand butt. The models being tested are launched from a smooth-bored gun by means of a discarding sabot. The sabot protects the model from blast effects or damage while in the barrel as well as permitting the launching of odd-shaped combinations. It is of four petal construction; these petals opening and shearing due to air pressure after the sabot and model leave the gun (Figure 2). The sabots are designed so that the petals scatter rapidly from the line of fire. A large piece of armour plate, located at the blast wall and having a 9 in diameter hole cut in it, stops these petals from entering the range while letting the model continue unimpeded. In some cases, such as the launching of mortar projectiles, these sabots are not necessary.

Figure 2
Separation of model and sabot. M = 1.5 approx

The launching guns currently in use consist of smooth-bored 17-pounder and 5.5 in barrels, having internal diameters of 3.125 in and 5.9 in respectively. These barrels enable launching of fairly large scale models. To date some of the models tested have had wing spans of 5.75 in and body lengths of the order of 2 ft. Launching velocities extend from 600 to 4,000 ft/sec. A "Hi-lo" technique is used in burning the propellant. This technique distributes the gas pressure over a longer period of time, giving the same total thrust but avoiding the high peak pressures obtained in the normal chambers. Thus the acceleration that the models are subjected to is considerably lower than would be experienced using standard internal ballistic methods.

Inside the range (Figure 1) light screens are placed every 50 ft along the tunnel axis to measure velocity and at the 22.5 ft station a parallel-beam spark shadowgraph-schlieren system, employing 36 in diameter, optically ground (¼ wavelength of sodium light) paraboloidal mirrors, is located. The angular orientation of the model and its displacement from the line of fire are

[a]Proof and experimental firings of cannon have been carried out in the U.K. for more than 600 years and probably the oldest appointment in the Ordnance was made in the 13th Century when Roger de Leyburn was appointed Master Gunner of England. The duties of this office appear from an entry during the reign of Queen Elizabeth I in the Tower of London quarter books. They were 'to profess and teach his art to all under gunners in the exercise of shooting off great pieces of Ordnance'. This 'art' covered not only the service of the weapon in action but also its proof, and a book published about this time under the title of 'The Complete Souldier' advises 'Prove your guns with the weight of the iron bullet thereof, then with five-fourths, then with three halves' – probably the earliest 'proof specification' on record. The Headquarters of artillery activity at that time was the Tower of London. Here lived the Master Gunner of England with his fee'd gunners and scholars. The firing of cannon was carried out at the Old Artillery Gardens in Spitalfields and later at the New Artillery Gardens at Moorfields where the H.A.C. now have their Headquarters. Pepys records a visit in 1669 to 'Moorfields where I never was before but now, by Capt. Dean's invitation did go to see his new gun tryed, this being the place where Officers of the Ordnance do try all their great guns'.

* * *

In 1716 Col. Borgard was Colonel of the then newly-formed Regiment of Artillery; the Master Gunner of England had his title changed to Master Gunner of Whitehall and St. James Park and disappeared from the Woolwich scene. Borgard was appointed Assistant to the Surveyor General and was given a free hand where artillery equipments were concerned and took over their proof which during the first half of the 18th Century tended to become much more of a regimental and social affair, and it is recorded that Ordnance was proved before the Principal Officers of the Board of Ordnance (the new title of the Office of Ordnance) 'to which many of the nobility and gentry are often invited and afterwards sumptiously entertained by them'.

(Extract from '*A Note on the History of Proof*' from the MANUAL OF THE CANADIAN PROOF OFFICER'S COURSE.)

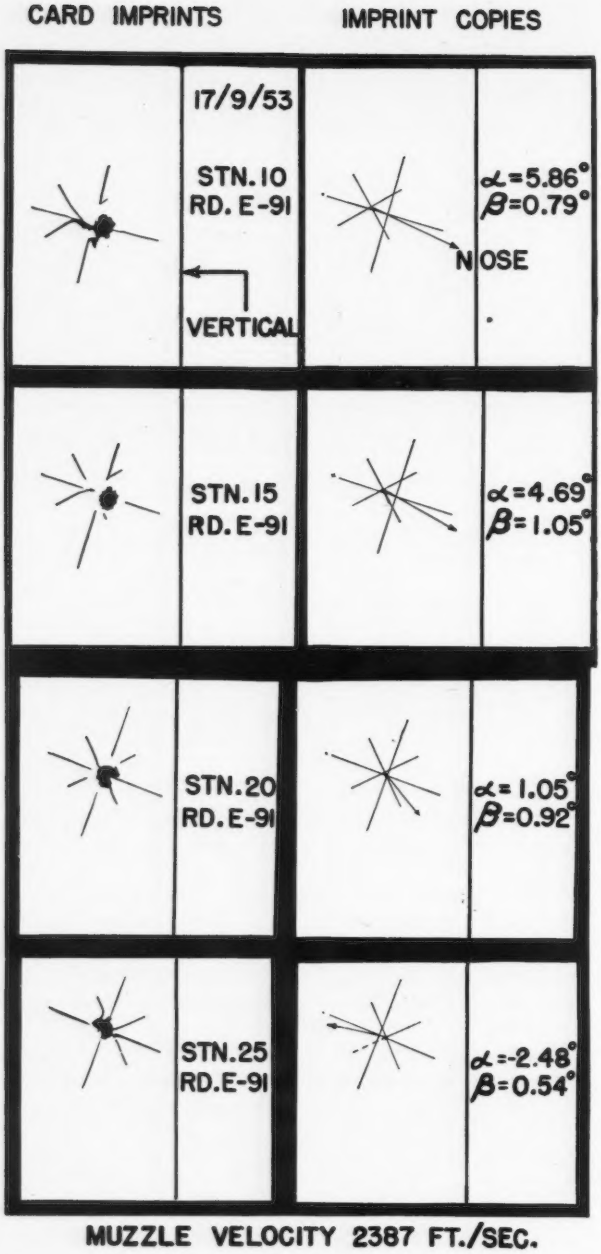


Figure 3
Yaw-card imprints

measured by means of yaw cards mounted in metal and wooden frames along the range axis; their spacing, size and distribution being flexible and determined by test requirements (the size of the cards increases proceeding down the range to allow for accidental as well as deliberate dispersion, cards as large as 9 ft × 12 ft being used). Each card has marked on it a horizontal and vertical line; the intersections of these lines define points lying on a horizontal line in the vertical plane of fire and passing through the gun trunnion. The angular offset of the gun barrel with respect to this reference line is measured for each test. The cards themselves are held in their frames in such a manner as to prevent any initial waviness. A model passing through the card will leave a trace, the shape and size of which depends on the model's angular orientation with respect to the velocity vector. Only certain types of paper can be used if good model signatures are required. The paper found most suitable to date in the work reported herein is almost identical to normal blotting paper. Figure 3 shows the signatures left by a wing-body-tail model. The reduction of angles from these cards is relatively easy. To identify the roll orientation, one of the wing panels has a light coating of machinists' blue put on it just prior to firing. The blued panel is clearly identifiable on each card, even when as many as 75 cards are used. The angle of attack of the model may be found by measuring the relative displacement of any two model cross-sections. Usually the displacement of the tail with respect to the wings or nose is used to determine angle of attack and for shapes lacking fins, such as mortar bombs and gun shells, nose and base imprints are used. When care is taken, surprising accuracy can be achieved in these measurements.

For determination of aerodynamic characteristics, dynamically similar models are not required. However, if trajectory and, particularly, certain types of dynamic coupling studies are to be performed, dynamically similar models are needed. Such models present a more difficult design problem than the normal aerodynamic models since weight and balance requirements must be met despite the high acceleration loadings experienced in the barrel. In the work to date in the C.A.R.D.E. range no particular difficulty has been encountered in meeting the requirements of dynamic similarity of missile and other configurations tested.

Model construction for the range firings is one of the most difficult problems encountered. For missile and aircraft models launched from the 3.125 in barrel, it has been found necessary to machine the deflection onto the control surfaces. This raised the cost of the models considerably and, since recovery is impossible, was a serious objection to the technique. For models launched from the 5.9 in barrel, however, such as the wing-body combination illustrated in Figure 4, it has been found that the wings can be attached to the body by means of a shaft and set at any angle desired. After setting, the wings are silver-soldered into place. The models made in this manner are relatively cheap and easy to fabricate. Prior to firing the model, geometrical tolerances, mass, inertia and center of gravity position are measured. Air density, temperature and moisture content are measured for each test using normal techniques.

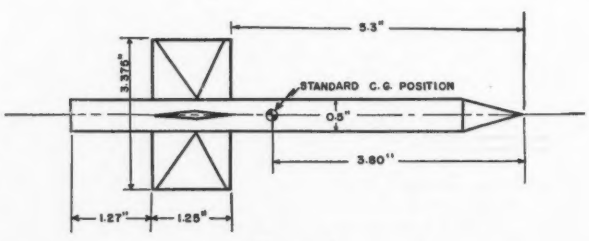


Figure 4
Cruciform wing-body combination tested in the Aeroballistics Range

LIST OF SYMBOLS

b — gross wing span (tip-to-tip)

c — wing chord

C_D — coefficient of drag based on gross wing area

C_l — total roll moment coefficient based on gross wing area bc and wing span b

C_{l_p} — damping in roll coefficient $= \partial C_l \Big/ \partial \frac{pb}{2V}$

C_{l_δ} — aileron power coefficient $= \partial C_l / \partial \delta$

C_M — pitching moment coefficient based on gross wing area and wing chord

C_{M_α} — rate of change of pitching moment coefficient with angle of attack

$C_{M_{\dot\alpha}}$ — $= \partial C_M \Big/ \partial \frac{\dot\alpha c}{2V}$ — rate of change of pitching moment coefficient due to the rate of change of angle of attack

C_{M_q} — $= \partial C_M \Big/ \partial \frac{qc}{2V}$ — damping coefficient

C_N — total normal force coefficient based on gross wing area bc

C_{N_α}, C_{N_δ} — rate of change of normal force coefficient with angle of attack and control deflection respectively

m — model mass

p, q, r — components of angular velocity of the model with respect to a triad fixed in space but instantaneously coincident with the body axes

s — distance along the flight trajectory

t — time

x, y, z — forms an orthogonal triad fixed in the body

$\hat{x}, \hat{y}, \hat{z}$ — forms an orthogonal triad fixed in the body in pitch and yaw but fixed in space in roll

P — rate of change of roll orientation with distance; degrees per foot

P_∞ — value of P when $\dot{P}$ is zero

M — Mach number

M_α — $= C_{M_\alpha} \rho V^2 Sc/2 =$ rate of change of moment about the pitch axis with angle of attack

$M_{\dot\alpha}$ — $= C_{M_{\dot\alpha}} (\rho V^2 Sc/2)(c/2V) =$ rate of change of moment about the pitch axis due to rate of change of angle of attack

M_q — $= C_{M_q} (\rho V^2 Sc/2)(c/2V) =$ rate of change of moment about the pitch axis due to rate of pitch

F — aerodynamic normal force $= C_N \rho V^2 S/2$

I, I_y, I_z — moment of inertia about the longitudinal axes, assumed in most of this work to be equal

I_x — moment of inertia about the axis of roll

S — reference area, usually gross wing area bc

V — model velocity

XYZ — components of an orthogonal triad fixed in space

α, β — angles of attack in pitch and yaw respectively referred to a body axes system

$\hat{\alpha}, \hat{\beta}$ — angles of attack in pitch and yaw respectively referred to a non-rolling axis system

α' — total angle of attack (angle between velocity vector and model center line)

$\hat{\Phi}$ — bank angle; angle between the horizontal plane and the plane containing α'

$\hat{\xi}$ — $= \hat{\beta} + i\hat{\alpha}$

δ — the amount of differential deflection of a pair of planar wing panels from the mean

δ_p — control deflection producing a moment about the pitch axis; positive if it produces a force in the positive sense of z

δ_y — control deflection producing a moment about the yaw axis; positive if it produces a force in the positive sense of y. (Note that with this definition of δ_y and δ_p,

$$F_{\delta_p} = F_{\delta_y}, \quad -M_{\delta_p} = +M_{\delta_y})$$

δ_ϵ — $=$ resultant control deflection $= \delta_y + i\delta_p$

ρ — air density

AR — aspect ratio (gross) of the wings

B' — $\sqrt{M^2 - 1}$

SOLUTIONS TO THE EQUATIONS OF MOTION

The resultant spatial trajectory of any model can be related directly to the motion of full scale bodies satisfying the requirements of similitude. Thus the aeroballistics range technique can be used directly for trajectory studies and to predict trajectories. However, for determination of aerodynamic quantities the aeroballistic trajectories must have fitted to them the general equations of motion. While generally this fitting may best be accomplished on some form of automatic computing device, hand fitting known solutions to the experimental curves can be made to yield quite satisfactory results (all of the results presented in this paper were obtained in this manner).

The ballisticians were among the first technical groups to engage in applied studies of rigid body motion in space.[1, 2] The classical theory of aeronautical dynamics was advanced by Lanchester and modified by Bryan and others at the start of the present century. This theory described the trajectories of bodies under an aerodynamic force system when moving in a plane of symmetry. Inherent in this theory is the assumption that coupling between longitudinal and lateral degrees of freedom is negligible. The solutions to the equations for motion about any axis of longitudinal or lateral symmetry formed the basis of aircraft dynamic analysis up until recent times.

The use of rifled shells led to investigation in France, Great Britain and other European countries of the

motions of bodies with large roll rates. In 1920, Fowler, Gallop, Lock and Richmond[3] set up and solved the equations of a spinning shell accounting for aerodynamic normal force, pitching moment, damping and magnus terms. They were able to perform some experimental trajectory checks on their theory.

During the Second World War, German scientists engaged on rocket and fin-stabilized shell programmes observed that oscillations of cork-screw like character and frequent sudden instabilities occurred when moderately low roll rates were present. Schneller[4] postulated a theory to account for such motion by the introduction of an aerodynamic force normal to the plane of resultant angle of attack. This theory is generally unsatisfactory and does not explain the basis of the introduced lateral force for bodies with trigonal or greater symmetry. After the war, the free-flight section of the Ballistics Research Laboratory,[5, 6] by use of approximations valid for a large class of finned bodies possessing trigonal or greater symmetry, succeeded in reducing the equations of combined pitch, yaw and rolling motion for the case of constant roll rate and flight velocity to a second order, non-homogeneous, linear ordinary differential equation with constant coefficients. In this approach an axis system (first introduced in Reference 3), fixed in the body in pitch and yaw with the x-axis lying along the body center-line but fixed in space in roll, is used and

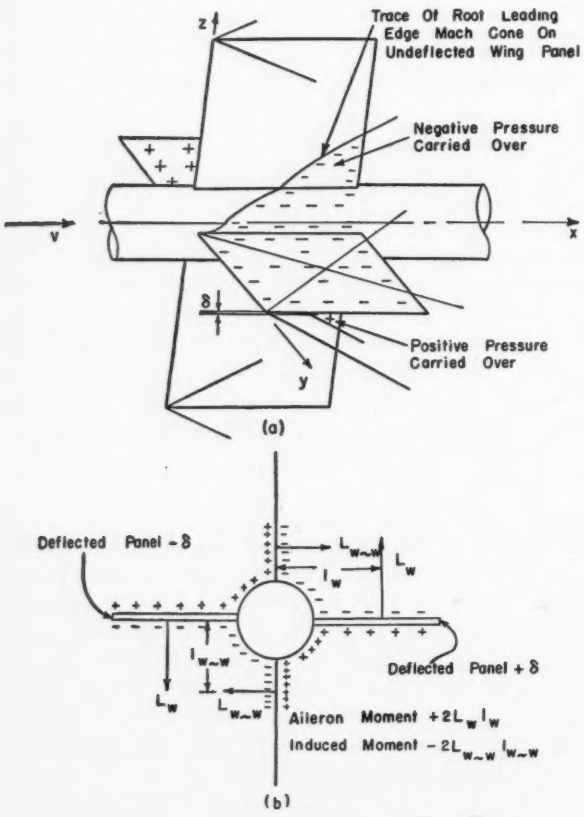


Figure 5
Induced rolling moment due to complementary panel interference

quantities measured with respect to such a reference system are denoted by placing a bar on top of them, $\hat{x}$, $\hat{y}$, and $\hat{z}$. The solution to these equations was shown[5, 6] to take the form:

$$\hat{\xi} = K_1\epsilon^{(\lambda_1 + i\omega_1)t} + K_2\epsilon^{(\lambda_2 + i\omega_2)t} + K_3\epsilon^{ipt} \qquad (1)$$

where $\hat{\xi} = \hat{\beta} + i\hat{\alpha}$, K_1, K_2, K_3 are in general complex quantities, while λ_1, λ_2, ω_1, ω_2 are real, and p is the roll rate and is assumed along with the velocity, to be constant. For statically stable vehicles, it was shown:

$$\lambda_{1,2} = \frac{F_\alpha}{2mV}(1 \mp \tau) + (M_{\dot{\alpha}} + M_q)\left(\frac{1 \pm \tau}{2I}\right) \qquad (2a)$$

$$\omega_{1,2} = \frac{pI_x}{2I}(1 \pm 1/\tau) \qquad (2b)$$

where $$\tau = 1 \Big/ \sqrt{1 - \frac{4IM_\alpha}{p^2 I_x^2}}$$

From these results, it is seen that the complex angle of attack is described by three rotating vectors, two of which are damping. By determining the rotational velocity of each vector, the pitching moment can be derived and by measuring the damping of each vector, the aerodynamic damping derivative can be found. The method of reduction using Eq. (1) will be described later.

RESULTS FOR A CRUCIFORM, RECTANGULAR WING BODY COMBINATION

Nine models of the wing-body combination, shown in Figure 4, were fired in the aeroballistics range at supersonic Mach numbers between 1 and 2. These models had varying amounts of differential deflection of one pair of planar wing panels, but no resultant control settings. Launching introduced oscillatory angles of attack as large as 4 degrees into the model flights. Various center of gravity positions were used, five models tested with the center of gravity in the standard position (Figure 4) and four models with the center of gravity in positions fore and aft of this point. For most of the tests, cards were placed in position along the tunnel axis at 5 ft intervals over approximately the first 140 ft and at irregular intervals thereafter. For two tests, in order to assess card interference effects, only seven yaw cards, located at stations 25, 30, 35, 160, 170, 180 and 190 ft, were used.

Roll Results

Under the assumption that the wing-body combination tested is unaffected in its roll rate by small angles of attack, an assumption verified by wind tunnel tests, and assuming a constant flight velocity, it can be shown that the rate of change of roll angle with distance is given by the equations:

$$P = -\frac{B}{A}(1 - \epsilon^{As}) \qquad (3)$$

$$P_\infty = -\frac{B}{A} \qquad (4)$$

where the boundary conditions assumed are: $P = 0$ at $s = 0$ and $P = P_\infty$ ($\dot{P} = 0$) when s is large. The

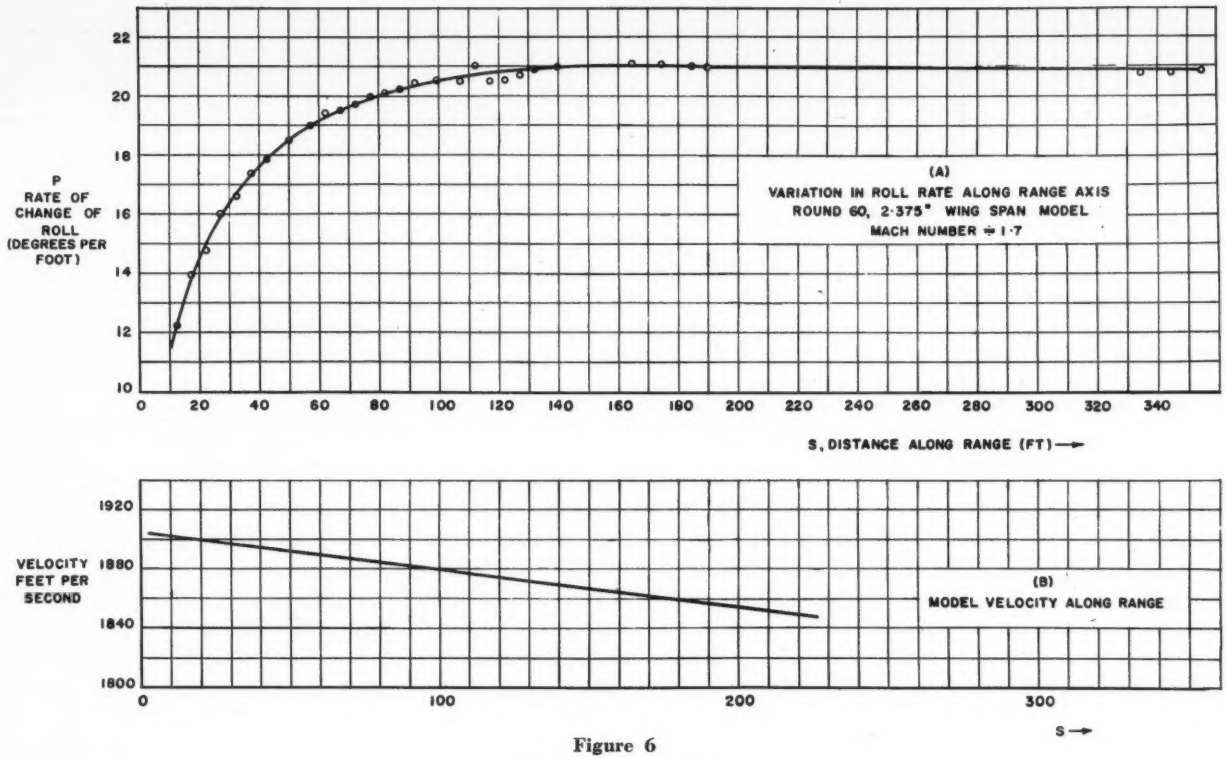


Figure 6
Build up in roll rate

quantities A and B are damping and aileron power terms respectively, defined by the relations:

$$A = \frac{\rho S b^2}{4 I_x} (C_{l_{p_w}} + C_{l_{p_{w\frown w}}}) \tag{5}$$

$$B = \frac{\rho S b}{2 I_x} (C_{l_{\delta_w}} + C_{l_{\delta_{w\frown w}}}) \delta \tag{6}$$

where the aileron power is considered to be comprised of two components:

(i) the basic roll moment due to lift distribution on the deflected panels (the moment $2L_w l_w$ of Figure 5) and

(ii) the induced roll moment due to carry-over of velocity and pressure fields from a panel to the panels at 90 degrees from it (the moment $-2L_{w\frown w} l_{w\frown w}$ of Figure 5).

Similarly the damping is considered as due to:

(i) the damping due to incidence variation along the wing panel caused by the roll velocity and

(ii) the undamping due to panel carry over.

When Eqs. (3) and (4) are combined and logarithms taken, there results:

$$ln \frac{P_\infty - P}{P_\infty} = As \tag{7}$$

Hence a plot of $ln \frac{P_\infty - P}{P_\infty}$ versus distance should yield a straight line, the slope of which is A. Once A is determined, B may be calculated from Eq. (4) and $(C_{l_{p_w}} + C_{l_{p_{w\frown w}}})$ and $(C_{l_{\delta_w}} + C_{l_{\delta_{w\frown w}}})$ from Eqs. (5) and (6).

The build up in roll rate along the flight trajectory is shown for a typical model in Figure 6, the terminal roll rate is achieved at about the 150 ft position. The logarithmic plot for this round is presented in Figure 7, along with mass, inertia, reference area and length values, and the damping in roll coefficient derivative and aileron power coefficient thus determined are marked on this Figure.

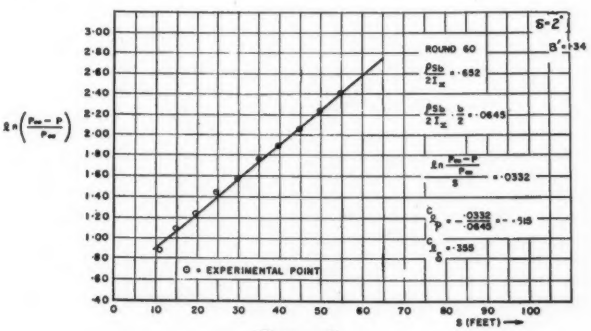


Figure 7
Logarithmic plot of roll rate

Figures 8 and 9 summarize the aileron power and damping in roll results obtained from the aeroballistics range tests on this configuration. Included on these plots are the theoretical curves obtained by applying linearized theory to the combination under the assumption that:

(i) the body is at zero angle of attack;

(ii) all interference effects other than wing-wing carry-over are negligible.

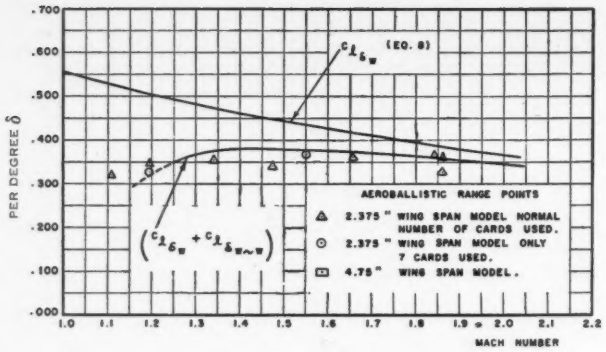

PER DEGREE δ
MACH NUMBER
(EQ. 8)
AEROBALLISTIC RANGE POINTS
2.375" WING SPAN MODEL NORMAL NUMBER OF CARDS USED.
2.375" WING SPAN MODEL ONLY 7 CARDS USED.
4.75" WING SPAN MODEL.

Figure 8

Theoretical and experimental aileron power coefficient cruciform rectangular wing-body combination tested with one planar pair of panels differentially deflected the other pair of panels being undeflected.

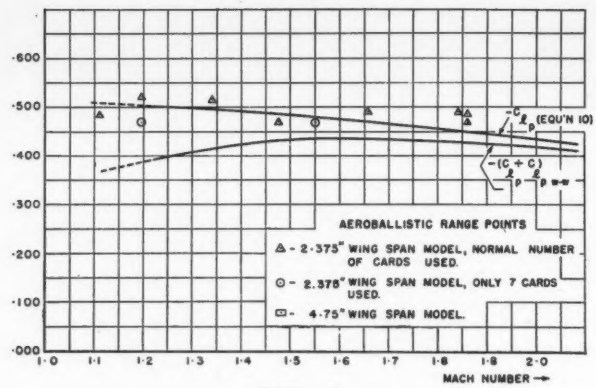

(EQU'N 10)
AEROBALLISTIC RANGE POINTS
Δ - 2·375" WING SPAN MODEL, NORMAL NUMBER OF CARDS USED.
⊙ - 2.375" WING SPAN MODEL, ONLY 7 CARDS USED.
⊡ - 4·75" WING SPAN MODEL.
MACH NUMBER

Figure 9

Theoretical and experimental damping in roll coefficients cruciform rectangular wing-body combination tested with one planar pair of panels differentially deflected the other pair of panels being undeflected.

By applying strip integration methods to the panel loadings, it was found that the aileron power when one planar pair of panels are differentially deflected is:

$$C_{l_{\delta_w}} = \frac{1}{B'}\left[1 - \frac{1}{AR\,B'} + \frac{1}{2\,AR^2\,B'^2} - \left(\frac{d}{b}\right)^2\right] \qquad (8)$$

where δ is the deflection of a panel from the mean. The effect of wing-wing carry-over onto the undeflected panels for wings in cruciform was found by a similar method to be:

$$C_{l_{\delta_{w\sim w}}} = -\frac{2(d/b)^2}{n\,B'}\left(\frac{n^3}{6\,B'^2} - n + \frac{2\sqrt{2}}{3}B'\right) \qquad (9)$$

where $n = \dfrac{\text{panel chord}}{\text{body radius}}$. From Figure 8 it is seen that the theoretical values of aileron power accounting for carry-over are in good agreement with the experimental measurements.

The damping in roll for the cruciform model ($d/b = 0.21$), with no panel-panel interference, was calculated by replacing the body by the immersed portions of the wing panels and using the formula derived in Reference 6.

$$C_{l_{p_w}} = -\frac{2}{B'}\left[\frac{2}{3} - \frac{1}{AR\,B'} + \frac{1}{3(AR\,B')^2} + \frac{1}{12(AR\,B')^3}\right] \qquad (10)$$

The effect of panel-panel interference on damping in roll was found by deriving an interference potential by integration of supersonic sources over the blanketed panel areas. This yields the result:

$$C_{l_{p_{w\sim w}}} = \frac{16}{3\pi\,B'^4\,AR^2} \qquad (11)$$

From Figure 9 it is seen that the experimentally determined damping in roll results agree quite well with the theoretical estimates using Eq. (10) only – that is, when interference effects are assumed negligible. This agreement could be due either to an over-estimation of interference effects by Eq. (11) or to the contribution of the cards to damping.

Effect of Yaw Cards on the Roll Results

In order to assess the effect of the cards used in the roll tests, some models were fired using only three cards to measure the roll rate and acceleration at one point during the build up phase (these cards were located at stations 25, 30 and 35) and four to measure terminal roll rate (located at stations 160, 170, 180 and 190). The locations of the cards to give terminal roll rate were determined from the results of previous tests and these cards were spaced over a sufficient length of the tunnel to allow for any moderate changes in terminal roll rate due to card effects.

The results for two such rounds are included in Figures 8 and 9 and it is to be observed that they are indistinguishable from the remainder of the experimental results. Generally the results obtained are reliable to within 5 per cent.

Pitch-Yaw Results

Method

As previously described, the angle of attack variation in pitch and yaw as the model proceeds down-range is measured by means of yaw cards. When the angles of attack are referred to a non-rolling axis system, their variation along the trajectory should be described by Eq. (1) over that portion of the range where the velocity and roll rate may be assumed constant. A typical experimental plot of angle of attack variation under such conditions is presented in Figure 10 (Round 60).

The wing-body combinations tested possessed no control deflections to produce resultant pitch or yaw other than those due to manufacturing tolerances. Hence the trim vector K_3 must be small. On this basis the reduction of aerodynamic derivatives was performed in the following manner, an adaptation of the method suggested in Reference 5.

(i) The bank angle $\hat{\Phi}$ was plotted versus distance, and the slope of the best straight line taken as the first approximation to ω_1. For Round 60, the slope was 10.2° per ft.

(ii) Each experimental point is now rotated back through an angle $\omega_1 s$, where the assumed value of ω_1 is used and where s is measured from the first experimental point. If the estimate of ω_1 is exact, we see from Eqs. (1) and (2) that the K_1 vector will cease rotating and the K_2 and K_3 vectors will rotate with angular velocities $(-\omega_1 + \omega_2)$ and $(-\omega_1 + p)$ about the end of the K_1 vector. Since K_3 is small, the

resultant figure will be close to a circular spiral; since K_1 is damping, the center of this spiral will be moving towards the origin. This method is extremely sensitive to any error in ω_1; in the example, an error of .03° per ft could be detected; it was found that the value for ω_1 of 10.2° per ft, assumed from step (i), was too large and a value of 10.0° per ft was found to be satisfactory. The resultant spiral for the case of Round 60 is included in Figure 10.

(iii) Having determined ω_1, the pitching moment coefficient can be calculated from Eq. (2b) using the measured roll rate.

(iv) The centers of successive portions of the spiral are estimated; this is done by assuming that over a

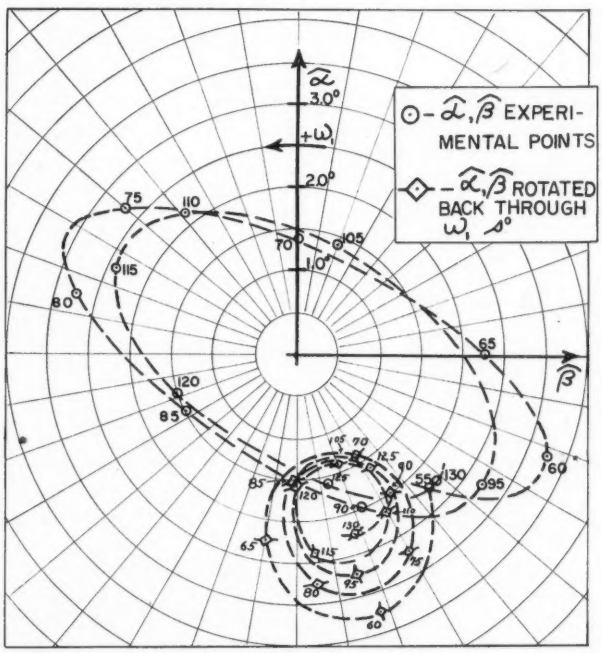


Figure 10
Experimental result from Aeroballistics Range for Round 60 Mach number 1.7

small portion of the flight trajectory, 15 ft in this case, the damping of the K_1 vector is negligible. The vector from the origin to the estimated center is then the K_1 vector and its variation with distance is then determined. By plotting logarithmically the amplitude of this vector versus position, using the mid-point of the elementary spiral length, the damping factor, λ_1 may be determined. This plot for Round 60 is shown on Figure 11a.

(v) The K_1 vector is subtracted from the rotated experimental points by translation of the center of each successive branch of the spiral to the origin.

(vi) With the K_1 vector removed, the points are now rotated about the origin through an amount $(\omega_1 - p)s$, so that there should result a stationary constant K_3 vector with K_2 damping and rotating

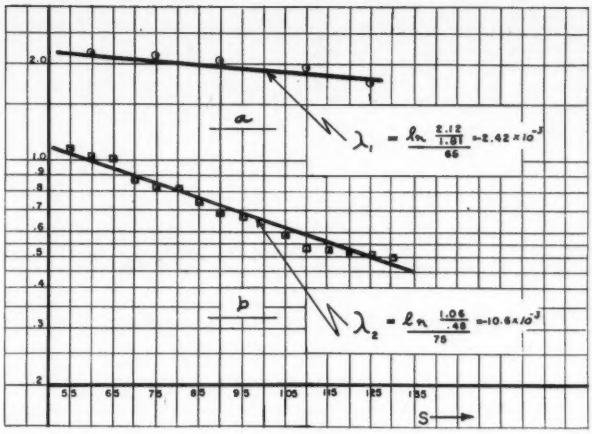


Figure 11
Logarithmic plot of amplitude of K_1 and K_2 vectors versus distance, Round 60

about the end of this vector. By determining the center of the resulting spiral, the K_3 vector may be determined. For Round 60 the resulting spiral is presented in Figure 12, and is centered very nearly on the origin, so that K_3 is approximately zero.

(vii) The K_3 vector thus determined is then subtracted from the original experimental points and the process repeated. Successive iterations may be carried out until the desired accuracy is obtained. The K_2 vector is determined once the K_1 and K_3 vectors are removed and by plotting the amplitude of K_2 versus distance on semi-log paper, λ_2 is determined. (This plot for Round 60 is shown in Figure 11b) The angular velocity of ω_2 is obtained from a plot of phase angle versus distance for the K_2 vector.

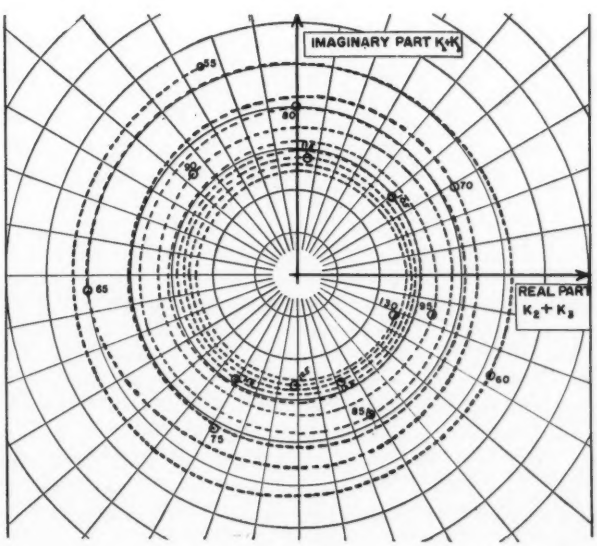


Figure 12
Polar plot of $K_2 + K_3$ for Round 60, K_3 estimated to be negligible

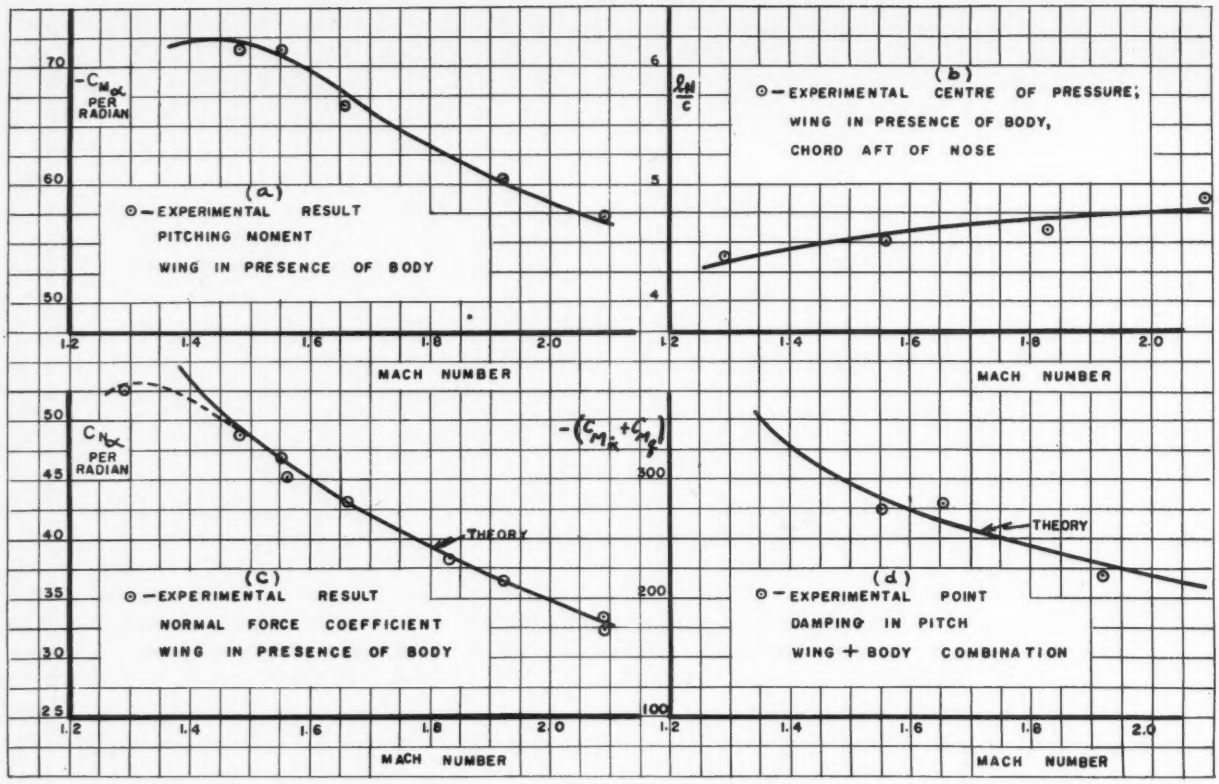


Figure 13
Experimental result for wing-body combination

Experimental Results

Figure 13a presents the variation with Mach number of the pitching moment coefficient for the wing in presence of the body as determined experimentally from rounds with their center of gravity in the standard position. In this plot, the body-nose moment corresponding to the R.M.S. value of angle of attack has been subtracted from the experimental $C_{M\alpha}$ for the wing-body combination.

Figure 13b presents the variation with Mach number of the center of pressure of the wing in presence of the body obtained by combining the experimental pitching moment coefficient results for the different center of gravity positions. A scatter of some 5% from the mean curve may be observed; this can be improved by increasing the distance between fore and aft center of gravity positions.

Figure 13c presents the variation with Mach number of the normal force coefficient slope, obtained by combining the experimental pitching moment coefficient slopes with the faired center of pressure curve. The theoretical estimate of normal force using standard N.A.C.A. theory for estimating carry-over is included on this Figure. The experimental results may be observed to be in agreement with the theory.

Figure 13d presents the variation with Mach number of the damping in pitch derivative for the standard center of gravity position. For two rounds, the amplitude of the angular oscillations was too small (less than 1°) to permit accurate determination of this quantity. The presented experimental results are seen to agree fairly well with the theoretical estimates, made by application of the methods of Reference 7.

Drag Results

The total drag coefficient of the rounds is obtained by measuring the retardation. The drag due to incidence is determined from firing non-rolling rounds with varying amplitudes of angle of attack oscillation and plotting the total measured drag versus the square of the R.M.S. angle of attack. The slope of the straight line thus obtained yields the drag coefficient due to incidence, whilst the intercept yields the zero-lift drag coefficient. For rounds which are rolling as well as pitching and yawing, such as the rounds considered here, the drag due to lift must either be determined from non-rolling range tests or from wind tunnel measurements. The latter results were used here; once the drag due to incidence was removed, the drag due to roll rate was determined by plotting the remaining drag coefficient versus rate of roll for constant Mach number. The contribution of roll rate to drag coefficient was then removed and the zero lift drag values determined. These drag results for the rounds tested are presented in Figure 14.

GENERAL STUDIES IN THE RANGE

Control Effectiveness

The general method of data reduction previously described may be adapted to the determination of control effectiveness for wing-body-tail configurations. A typical wing-body-tail combination tested in the aeroballistics range is shown in flight in Figure 15; the wing-body combination is geometrically similar to that shown in Figure 4, but the wings have been moved forward and a narrow delta cruciform tail indexed at 45° with respect to the wings has been added at the rear. All surfaces have zero deflection except one planar pair which have a

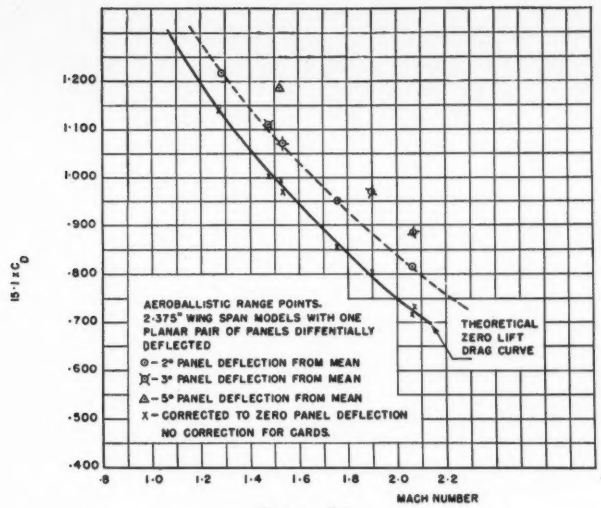


Figure 14

Drag variation with Mach number, cruciform rectangular wing-body combination

pitch deflection of 3°; the model attained a roll rate of approximately 0.5° per ft, due to tolerance misalignment etc.

For accurate results, the K_1, K_2 and K_3 vectors must be determined by the reduction procedure outlined. However, for small rates of roll, the trim characteristics can be determined fairly closely by consideration of the angle of attack variation in pitch and yaw. For the round shown these plots are presented in Figure 16. Under the assumptions of linearity, the ratio α_{trim}/δ can be combined with the known value of pitching moment coefficient to yield the pitching moment due to control deflection.

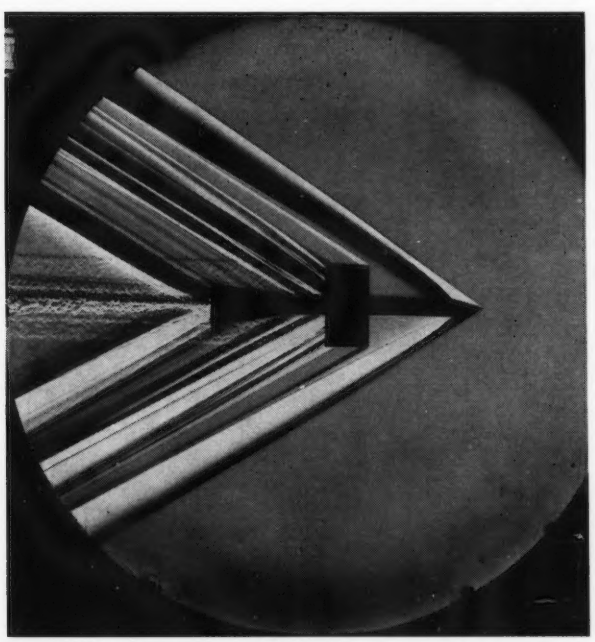


Figure 15

Wing-body tail combination in flight at Mach 2.0 with 3° resultant pitch deflection of forward wings

Standard reduction will yield the other aerodynamic coefficients as previously discussed.

Resonance

Theory indicates that when the roll rate is initially increased from a zero value, the trim vector is amplified and rotated out of the plane of the normal force due to control action. The amplification increases sharply when the roll frequency approaches the circular frequency in pitch and yaw and attains a maximum when these frequencies are equal. This phenomenon is described as resonance and is readily studied in the aeroballistics range. The results of some recent studies in the C.A.R.D.E. range are presented in Reference 8.

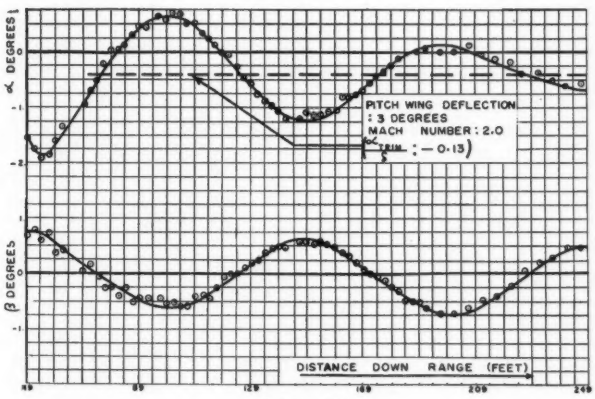


Figure 16

Oscillation in pitch and yaw wing-body tail combination

CONCLUSION

The aeroballistics range technique can be used to determine the standard aerodynamic quantities quite accurately. The use of yaw cards has proved satisfactory for the general shapes, which have included mono-wing aeroplane-like configurations, tested to date. The results presented in this paper are considered to be illustrative of the type and quality of data obtainable.

ACKNOWLEDGMENT

The author is indebted to all the members of the Ballistics Wing who co-operated in preparing the experimental data. He particularly wishes to acknowledge the assistance of Mr. G. H. Tidy in supervising the reduction of many of the rounds.

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AN INTRODUCTION TO AEROELASTICITY†

by B. J. Kaganov*

Canadair Limited

INTRODUCTION

THIS paper will be kept as non-technical as possible. The main theme will be restricted to defining some of the various aeroelastic problems the present day aircraft Engineering Department has to cope with. The scope of the subject is such that any of the engineers in the Dynamics Department can speak on it for days and fill the four walls of an average lecture hall with enough integral and differential equations to warm the hearts of college mathematics professors. It is beyond the scope of this paper to involve myself in the highly technical aspects of aeroelasticity and I will therefore, try to keep it as simple as possible.

The first important function of any Engineering Organization is to design an aircraft or missile to meet a set specification. This specification will contain certain requirements for performance and manoeuvreability and ensure sufficient strength to withstand any aerodynamic loads which may be encountered during take off, landing and flight.

Aeroelasticity, which includes problems in flutter, aileron and wing divergence, did not rear its ugly head until the early stages of World War II. Prior to that time, most aircraft structures, particularly wings, were designed for normal manoeuvring or gust g loads and landing loads. This resulted in rigid structures, since design speeds were relatively small. With the advent of higher forward speeds, even though manoeuvring g's remained about the same, the wings became relatively more flexible. Forward velocities were now beginning to approach critical speeds where aeroelastic effects had to be considered.

FLUTTER

One of the most common aeroelastic problems and the first to be considered seriously was flutter.

The phenomenon of flutter presents a problem in stability. Flutter is usually classed as an oscillatory instability, where the flexibility of the structure plays an important part. An airplane wing sitting on the ground, if disturbed from its equilibrium position, will have damped vibrations. The damping results from the internal friction of the materials. If, however, the wing is moving forward at a constant forward velocity and is disturbed from its equilibrium position, the induced vibrations may be damped, remain constant or diverge, as shown in Figure 1. The critical flutter speed may then be defined as the speed where the system is neutrally stable and the vibrations remain constant, for at this speed the airplane becomes unmanageable and any increase in speed will cause the vibrations to diverge. Flutter is a vibration in which energy is extracted from the airstream to help build up the amplitude, and a catastrophic failure can easily occur within a second of its start.

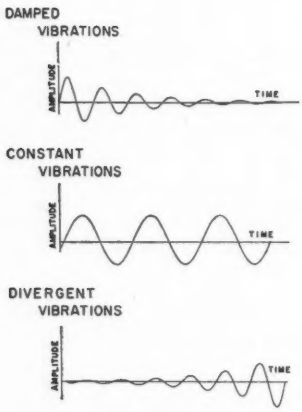


Figure 1
Induced vibrations

Modern high-speed aircraft are subjected to many different types of flutter and the flutter problem has a profound effect on their design. The means of curing flutter troubles usually involve increased stiffness or changes in mass distribution.

A particularly simple example of flutter is wing bending aileron flutter resulting from mass unbalance of the aileron. Suppose that a heavy mass is placed at the trailing edge of the aileron as shown in Figure 2. If the motion of the wing is upwards, the inertia of the mass tends to create a fixed point at the control surface trailing edge. The still air vibration of the control surface would be in phase with the wing bending, but the presence of aerodynamic hinge moments causes it to lag behind somewhat as shown here. The control surface

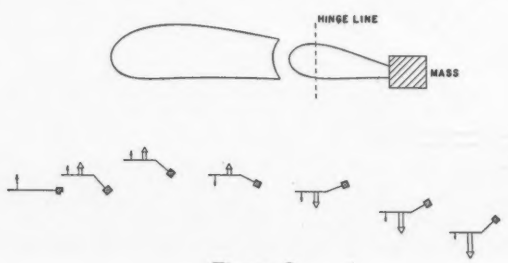


Figure 2
Aileron flutter

†Paper read before the Montreal Branch of the C.A.I. on the 15th February, 1956.
*Chief Structures Engineer.

deflects so as to produce a lift that feeds a net amount of work per cycle into the vibration and increases its amplitude. This type of flutter can, of course, be eliminated by putting the centre of gravity of the aileron at the hinge.

A conventional aircraft wing has the same vibration characteristics as a free-free beam. It is generally a beam tapered at each end symmetrically and supported at mid span to another beam, the fuselage, with stiffer characteristics. Wing resonances can be obtained either by calculation or measurements. Present practice is to calculate them during the design stage and then measure them prior to first flight. The resulting modes and frequencies are used to predict flutter speeds. The calculation of the vibrations of a beam (wing) leads naturally to an integral equation or alternatively a differential equation, but since this is beyond the scope of this paper, it will suffice to say that the dynamics engineers, with their tools of automatic electronic digital and analogue computers, spend many hours determining the modes and frequencies which are then used with aerodynamic parameters to obtain the critical flutter speed.

It may be opportune at this time to consider the sort of modes and frequencies obtained on an actual aircraft. The modes fall into two broad classes – symmetric and anti-symmetric. In any symmetric mode, the motion of one side of the centre line will be the mirror image, in this plane, of the motion on the other side; in an anti-symmetric mode the motion will be the exact reverse of this, as shown in Figures 3 and 4.

Almost always the fundamental symmetric mode of the aircraft is primarily one of wing bending, such as shown in Figure 3. The wing bends as a free-free beam with the centre section and fuselage moving down while the wing tips are moving up through a much greater amplitude. The mode with the next higher frequency will depend on the aircraft. It may be the wing torsion mode combined with wing bending or it may be fuselage vertical bending. If the aircraft is a small type, like

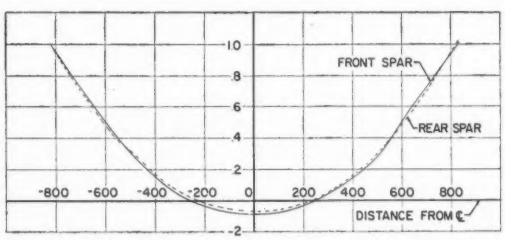


Figure 3
Aircraft wing symmetric deflection curve 2.8 cps

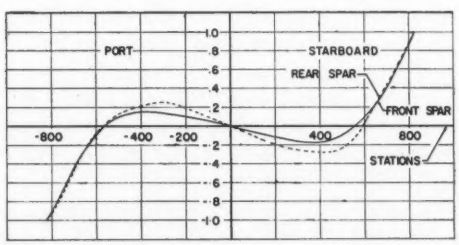


Figure 4
Aircraft wing anti-symmetric deflection curve 6.2 cps

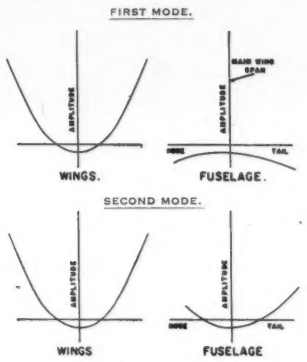


Figure 5
Modes of vibration

a fighter with a conventional fuselage and tail unit, and if this frequency is close to that of the wing fundamental, a pair of modes will exist with little frequency separation and considerable amplitudes of wing and fuselage bending in both.

In Figure 5 the first mode shows the wing amplitude plotted from tip to tip and it can be seen that there are two nodal points, one on either side of the fuselage, which in itself is bending in opposition to the wing. In the next mode, which will have a frequency approximately 25% higher than the first, the wings and fuselage bend in the same sense, with two nodes in each and the whole centre section moving in opposition to the extremities. These modal shapes are quite logical. In the first, the fuselage is being forced to vibrate below its own natural frequency and therefore does so in phase with the exciting force, in this case the reaction from the wing. The second frequency will be higher than the natural frequency of either the wing or fuselage alone, so that both have nodes.

The anti-symmetric resonances are similar to the symmetric in general character, except that fuselage vertical bending is now replaced by fuselage lateral bending and torsion, and the fin modes are introduced.

The first anti-symmetric wing bending mode will occur at a higher frequency than the symmetric fundamental mode.

Ground Vibration Testing

In order to insure the safety of the airplanes – in a manner equivalent to structural testing to prove the stress analysis – the calculated resonance properties are checked on the ground prior to first flight, by direct measurement of the resonances. The aircraft is supported in the most flexible manner available. In the case of large airplanes, this in itself becomes a problem. In normal size airplanes, where the natural frequency of the wing is greater than 5 cycles per second, the airplane may be supported on its own landing gear with the tires deflated. In an airplane the size of the CL-28, where the wing natural frequency is calculated to be about 2 cps, a more flexible system is required. The method devised by the test group of the Canadair Dynamics Department is to support the airplane at the wing jack points and nose jacks. Huge truck tires are mounted on nitrogen gas cylinders. One side of the rim is attached to the jack point and the other side to the gas cylinder.

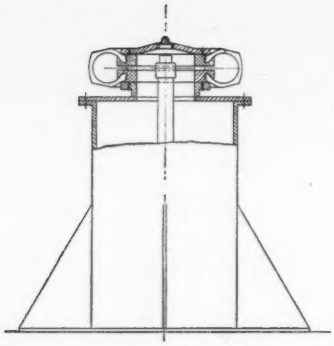


Figure 6
Detail of wing support

This type of support, illustrated in Figure 6, will allow slight movement and have a natural frequency of ½ cps.

The airplane is caused to vibrate by the attachment of electro-magnetic vibrators, and the amplitudes of response at a succession of frequencies are measured at selected points by means of electronic recording apparatus. For each of these points the amplitude can be plotted against frequency, as shown in Figure 7, and the resonances picked off. Each resonance frequency is then investigated in greater detail and amplitudes are measured over the whole aircraft so that modal shape as well as the frequency is obtained. On the CL-28 ground resonance tests it is planned to obtain the following data:

(a) Natural frequencies of the aircraft vertically and laterally.
(b) Relative modes of deformation and the nodal lines for each natural frequency.
(c) Degree of damping at each natural frequency.
(d) Natural frequencies and degree of damping of tabs.
(e) Investigation of abnormal vibrations found during the test.

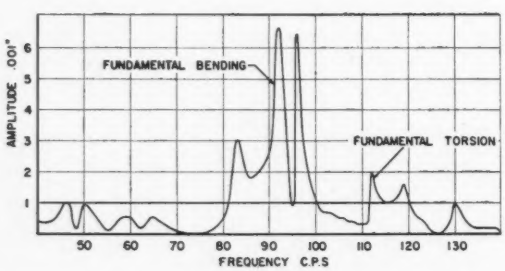


Figure 7
Frequency amplitude scan on missile wing

It is well to point out that resonance testing can be done efficiently only with the aid of theoretical analysis. Analysis to determine the coupled modes together with their frequencies will indicate the best possible locations for attaching the low frequency supports and vibration exciters; this is of the utmost importance when exciting the aircraft in its correct bending, torsion, symmetric and anti-symmetric modes. Also, as shown before, theoretical analysis will give the criteria for the design of the low frequency supports, to provide sufficient separation in frequency between rigid body deflection of the aircraft on its supports and the fundamental flexural vibration frequency of the structure. If perchance, in a plot such as shown in Figure 8, it is found that the measured fundamental frequencies vary greatly from the calculated, the measured frequencies are cranked back into the flutter formulae and the effects determined. The circuit is illustrated in Figure 9.

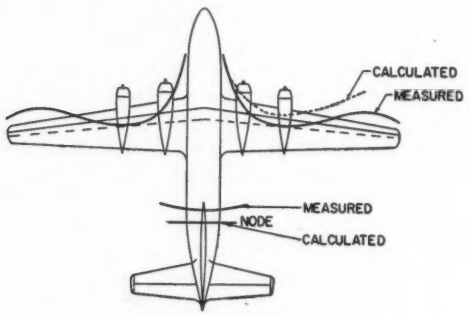


Figure 8
Nodal plot of aircraft 4.1 cps

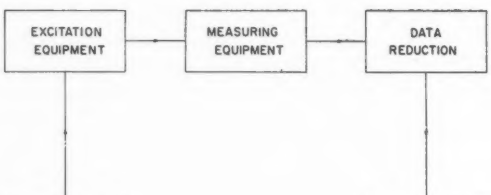


Figure 9
Ground testing block diagram

DIVERGENCE

All of us who have flown in modern transport type aircraft have noticed the deflections of the wings. During high speed flights, these deflections tend to redistribute the air loads and may cause their distribution to be significantly different from those computed on the assumption of complete rigidity. It is necessary that the structural members be designed for an airload distribution correlating to the structure when deformed. Consider the case of a normal wing with no sweep back and the elastic axis behind the aerodynamic center. The torque about the elastic axis due to lift tends to increase the angle of attack, which gives a greater lift and therefore again increases the torsional moment. If the additional increments of twist become smaller and reach a stable equilibrium, the design speed is then below the critical speed known as the divergence speed.

If conversely the torsional moments keep increasing, the wing will quickly reach a state where it will be destroyed. The problem of finding the load distribution on a straight wing is relatively simple. All one needs to do is calculate the torsional deflections.

The same process is found in a swept back wing, but in this case the deformations are usually favorable. Because of sweepback, torsional moments induce bending moments. With a swept wing the bending moment tends

to decrease the angle of attack and consequently the lift.

There are other types of divergence than wing divergence. One is aileron divergence which may be due to aerodynamic over-balance. It is common practice in the design of ailerons to provide some area forward of the hinge line, so that, when the aileron is deflected, the aerodynamic pressure on this area tends to increase the aileron angle, thereby helping to balance the aerodynamic restoring moments which oppose the pilot.

The net of aerodynamic restoring moment acts in opposite direction to the inherent structural stiffness of the control system. If at some speed the aerodynamic moment, which is dependent on the aerodynamic pressure, which in turn is proportional to velocity squared, overcomes the structural stiffness, the aileron divergence speed is attained and the control will flop over.

Similarly to wing divergence in high speed aircraft, tail divergence may also take place, with the rudder playing the important part in fin divergence and the elevator in stabilizer divergence. It is interesting to note at this point that divergence in itself rarely leads to failure of the part by direct action, since the secondary effects of a near-divergent condition would probably cause a premature failure. This is analagous to the Euler buckling load of a strut. Let us for example consider the wing torsional divergence. Approaching the divergence speed, heavy additional bending loads will be imposed on the wing because of the change of angle of incidence along the wing. More serious yet would be the effect of dynamic longitudinal stability in the form of sudden reduction in stick force per g. As an illustration of this, suppose a pilot applies up elevator angle to pull out of a dive at 2 g. This means that the basic wing lift force will be doubled but, if the speed is near the wing divergence, considerable nose-up wing twist will occur, which will increase the wing lift beyond what the pilot expects; the pullout may therefore be at 4 g rather than 2 g.

There are methods of curing divergence; the most costly weightwise is by increasing the torsional stiffness of the wing. Another is to have the flexural axis ahead of the aerodynamic center or, as shown before, sweep back the wings: see Figure 10.

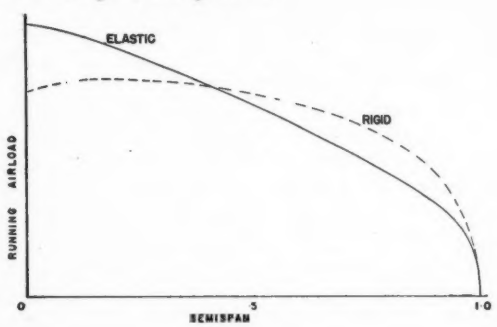


Figure 10
Airload distribution for rigid and elastic swept wing

AILERON REVERSAL

Divergence and flutter are problems of aeroelastic stability and at speeds beyond the critical speed a structural failure will occur. This is not true with aileron

Figure 11
Effect of speed on aileron

reversal which is a condition of no response to applied aileron. When the ailerons are deflected on an assumed rigid wing, the combined lift of the wing and aileron is increased on the side with the down aileron and increased on the side with the up aileron, together applying a rolling moment to the aircraft. With an elastic wing, the down aileron produces a torque about the elastic axis, which twists the wing and reduces the angle of attack of the wing, and the up aileron produces a torque which increases the angle of attack. Thus, it can be seen that wing twist acts in an opposite sense to aileron deflections. As the airplane speeds increase, torques caused by the ailerons twisting the wings are increased proportionally to the square of velocity, while the elastic restoring moment remains constant. Therefore, as higher speeds are attained, the rolling moment becomes less until a point where deflection of the ailerons will produce no rolling moment. This is known as aileron reversal speed. At speeds beyond the reversal speed, the aileron twists the wing to such a degree that loss in lift due to wing twist is larger than gain from aileron rotation and the aircraft rolls in the opposite direction. Figure 11 illustrates how aileron effectiveness, which is usually measured by the ratio of rolling velocity to aileron angle, is affected by speed.

SUMMARY OF CRITICAL SPEEDS

We have now seen three critical speeds that affect the design of the wing, namely flutter speed, divergence speed and aileron reversal speed. A comparison of their relative values is a necessary process during the early design stages. For a straight unswept wing of conventional construction, wing torsional divergence usually occurs at a considerably higher speed than aileron reversal, which in turn usually occurs at a higher speed than flutter. For swept back wings, the aileron reversal speed is lower than the flutter speed and the divergence speed is very high. This is illustrated in Figure 12.

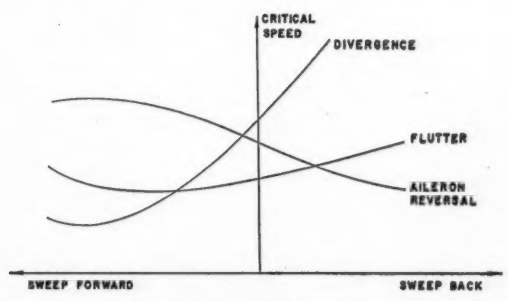


Figure 12
Comparison of divergence, flutter and aileron reversal speeds

LANDING AND GUST

The latest addition to the aeroelastic problems which confront the aircraft designer is the effect of flexibility of the structure during landing or upon encountering a gust. Heretofore the landing problem and the gust problem were both analyzed assuming a rigid airplane structure. Some similarity exists in the actual analysis of both cases. I will discuss first the landing case.

Landing

Although design criteria set forth by both military and civil requirements were felt to be high and unrealistic, a number of failures began to appear during normal landings. In trying to determine the reason for these failures, it was discovered that dynamic overloading due to the high spin up and spring back drag loads was a big factor. Arbitrary design conditions were then set up which eliminated the problem of design of the gear itself.

The vertical reaction measured at the gear was found to be much lower than that recorded at either the outboard nacelles or external packages hung at the wing extremities. The torsional moments on the wing, due to the sudden braking or spin up, were dynamically increased and large bending moments on the outer panels were found to be induced by substantial dynamic g's at the wing tip tanks. Thus the treatment of these loads as if the wing were rigid could no longer be applied and to insure the structural integrity it was again necessary to account for a flexible structure. More precise analyses were essential. The wing is now analyzed as a flexible beam for landing conditions, actual bending and torsional stiffness being taken into account.

Gust

The gust problem is actually two fold. What happens when the aircraft meets a gust? Firstly we must determine the severity of the gust. This is a meteorological problem and also one in statistics. Statistics have been and are continually being gathered in the U.S. and U.K. on this subject, using airplanes fitted with V-G recorders and counters determining the number and intensity of each gust. This data is being recorded at different altitudes and different geographical locations. Incidentally, this data is also very important in determining the fatigue life of transport type aircraft, since most fatigue failures are the result of large numbers of cycles at low loads.

Since the statistical problem of the severity of gusts had to be left to government agencies and years of study, the aircraft designer turned his attention to the second point and to the effect of gusts on design loads, considering the structure flexible rather than rigid.

In predicting the dynamic loads upon the structure, when the aircraft enters a gust of known intensity and gradient, a realistic analysis must take account of the free motion of the craft in space. Initially the plane is in equilibrium when entering the gust; the motion is disturbed by aerodynamic forces induced by the gust. The degrees of freedom of the airplane must be chosen to describe the aircraft response. For a structurally rigid airplane striking a symmetrical gust, only pitch and vertical translation need be considered.

If structural deformations are of importance, degrees of freedom must be added to describe their contributions to the response. In the gust problem, it is important not to neglect certain modes. Modes which seem unimportant in the flutter problem may be of prime importance in the gust problem, since we are now interested in wing stresses rather than an estimation of aircraft response.

CONCLUSION

To appreciate the reasons for the increased interest in the dynamic load problem, attention must again be brought to the increased demands of high speed flight, as well as the use of presently available materials of construction. These are both factors which tend to produce airframes with great flexibility, which in turn result in important inertia loadings and structural deformations. As shown before, these loadings will further produce additional aerodynamic loadings. It is then entirely within reason to conclude that severe dynamic loads can be encountered in certain combinations of circumstances.

I have tried to outline the major problems confronting the present day dynamicist. There are numerous other investigations being made, notably in propeller vibrations, engine vibrations, control systems and, not the least, noise control and sound proofing. If I have seemed to pass lightly over some of the phases of aeroelasticity, it must be attributed to the scope of this paper and to the general complexity of the subject.

RECENT MATERIAL DEVELOPMENTS FOR AIRCRAFT GLAZING†

by W. W. Farr*

The Rohm & Haas Co.

EARLY DEVELOPMENT

Until about 1939, the only generally available transparent plastic materials were the cellulosics. These were used in aircraft windows and canopies while only glass was used for the windshields. A number of the greenhouse-type enclosures were glazed with cellulose-acetate in order to save weight and installation cost.

One of the early successful transparent plastic materials was developed by Rohm and Haas Co., and bears the well known trade name "Plexiglas". This is a general-purpose cast acrylic sheet (methyl methacrylate). As early as 1937, some of this material had been applied to aircraft glazing. The first production type airplane to use Plexiglas canopies was the Bell Airacuda, followed closely by the Severski P-35 and one of the early Grumman Navy airplanes. From 1937 to 1940, the use of Plexiglas was spreading across the aircraft industry and the production facilities of the parent company were greatly expanded to meet the ever increasing demand.

At the start of World War II the type of material available was known as Plexiglas I and later, in 1942, Plexiglas IA was the standard material. This was formulated to give good cementing and forming characteristics and was used almost exclusively during the remaining years of the war and many thousands of parts were made. The airplanes of that period were not pressurized and the speeds were such that high temperatures were not a problem.

In 1944, the B-29's were in full production and as these were pressurized, further demands were placed on the material. The gun-sighting blisters were spherical in shape and were originally made of monolithic material. Several of these ruptured in flight, causing actual loss of crew members. When the domes ruptured, the pressure inside the airplane blew the gunners out through the open port. In order to decrease the probability of complete failure, laminated Plexiglas was developed and installed in these sighting blisters. This laminated material consisted of an inner 0.150″ thick Plexiglas sheet, a 0.150″ layer of polyvinyl butyral, and a 0.150″ outer thickness of Plexiglas. The polyvinyl butyral was sufficiently strong to maintain pressure in the airplane if the Plexiglas face sheets were ruptured. The lamination was done by the glass companies such as Libby-Owens-Ford and Pittsburgh Plate Glass.

†Paper read before the Montreal Branch of the C.A.I. on the 18th January, 1956.

*Technical Sales Representative.

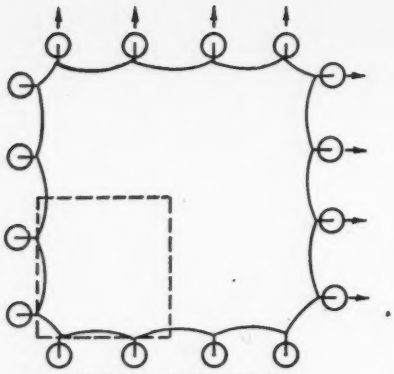
EARLY JET AGE DEVELOPMENT

At the end of World War II, the jet age was just coming into being. The availability of turbo-jet engines gave very much higher performance to aircraft and, therefore, imposed greater demands on the transparent enclosures. The jet planes were almost all pressurized and flew at speeds and altitudes where failure of the canopy might cause the loss of both pilot and aircraft. The higher heat resistance needed for jet-type aircraft, as well as a commercial demand for improved weathering resistance, led to the development of Plexiglas II. This material found a ready domestic market in that signs made of Plexiglas IA, when lighted and placed on southern exposure, sometimes developed temperatures as high as 190°F., which of course, was above the heat distortion temperature. Under these conditions the formed sections tended to revert to flat sheets, causing failure of the signs. Monolithic and laminated Plexiglas II have both been used for aircraft canopies since 1950.

RECENT DEVELOPMENT

More recently the increase in speeds possible for aircraft has gone up very rapidly, and speeds of Mach 1 to Mach 2 are actually realized in flight today. Under these conditions, a much higher heat resistant material becomes increasingly important as the heat rise due to adiabatic or pressure conditions on a standard summer day can be well above the heat distortion temperature of even Plexiglas II. To meet this requirement, Plexiglas 55, which is a modified acrylic sheet material, was developed and has been used successfully, and quite extensively, on these higher performance aircraft. Plexiglas 55 not only has a slightly higher heat distortion temperature, but has materially increased resistance to solvent and solvent-stress crazing. Because of this, the expected service life of glazing sections is materially increased over that previously available. For instance, on the DC-6 type transport airplanes, Plexiglas IA had a satisfactory average service life of approximately one year; Plexiglas II approximately two years; and from present indications, Plexiglas 55 should have two and one-half to three years life expectancy.

Another material developed by Rohm & Haas for aircraft glazing was Polymer "K". This material had very good heat and solvent resistance, but high notch sensitivity. It was supplied to the aircraft industry and sample installations were made on some training airplanes to determine its suitability for this use. After exhaustive testing, it was considered unsuitable for use in pressurized

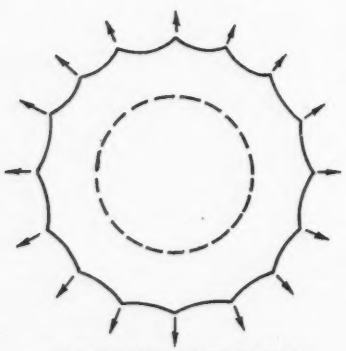


BIAXIAL STRETCHING

$$\frac{W_2 L_2}{W_1 L_1} = \frac{t_1}{t_2}$$

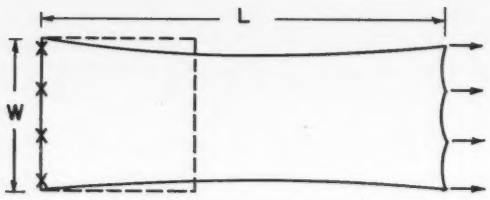
S_L = % Increase in L dimension = $\left(\frac{L_2}{L_1} - 1\right)$

S_W = % Increase in W dimension = $\left(\frac{W_2}{W_1} - 1\right)$



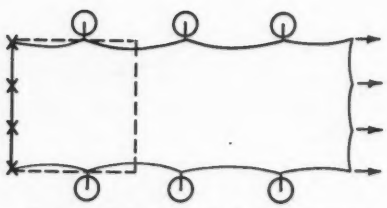
MULTIAXIAL STRETCHING

$$\% \text{ STRETCH} = 100\left[\sqrt{\frac{t_1}{t_2}} - 1\right]$$



UNIAXIAL STRETCHING

$$\% \text{ STRETCH} = 100\left[\left(\frac{t_1}{t_2}\right)^2 - 1\right]$$



RESTRAINED UNIAXIAL STRETCHING

$$\% \text{ STRETCH} = 100\left[\frac{t_1}{t_2} - 1\right]$$

Key t = thickness
L = length
W = width
S = stretch

Subscripts 1 and 2 denote conditions before and after stretch respectively.

Figure 1
Definitions relating to stretching

aircraft because of its comparatively high notch sensitivity, even though most of the other properties were excellent. The manufacture of this type material has been discontinued.

Several other firms in the United States and in Europe have been working on transparent plastic materials and two of these will be mentioned. The first is Sierracin, a polyester material which has been used commercially on the Martin 404's and some of the Constellation models' windows because of its craze resistance. This is a thermosetting rather than a thermoplastic material, and considerable difficulty has been experienced in trying to fabricate compound formed parts such as canopies. The notch sensitivity of polyester is also high as compared to the more recent Plexiglas developments, falling in the same general range as Polymer "K". Another material which seemed to give very great early promise was polymethyl alpha-chloroacrylate as made by Imperial Chemical Industries and the General Aniline & Film Co. This is believed to be a very difficult material to make in sheet form; and, although the properties of the material itself are interesting, the ability to produce it has not been sufficiently developed to make it commercially available.

Rohm & Haas is actively engaged in a research problem to develop new thermoplastic materials, which will be superior to those now commercially available, both in heat resistance and in resistance to stress and solvent-crazing. These materials are not ready for evaluation by the industry but progress has been encouraging.

STRETCHED PLEXIGLAS

Years ago, it was noticed that highly stretched Plexiglas sheets, exhibited some superior qualities over that of the "as cast" sheet. Notable among these qualities were toughness and craze resistance. Several articles and reports have been published discussing the investigation

and early development of the properties of stretched acrylic sheet. It was found that by stretching the material approximately 100%, the toughness would increase by about 5:1. At the same time the resistance to solvent and solvent-stress crazing would increase by a measurable amount. About three years ago, an investigation of the properties of multiaxially stretched Plexiglas was undertaken at Wright Air Development Center.

Prior to a discussion of the results from this and other research, it is necessary to define what is meant by the percentage of stretch. Figure 1 schematically shows the terminology used and gives formulae for the amount of stretch either uniaxially, biaxially, or multiaxially. While the information given is self-explanatory, in general, a biaxial stretch of 100% means an increase in the area of 4:1 and a decrease in the thickness of 1:4.

Stretching Methods

Along with the development of the material, it was necessary to develop machinery to obtain the stretched sheets and determine the proper conditions during the stretching operation which would produce the highest degree of improvement in quality. The first set-up which was used was a pressure head and clamping ring where a spherical bubble was blown and the material cooled in a hemispherical shape. This hemisphere was then transferred to a restraining device, the assembly placed in an oven and the temperature raised so that the central area of the hemisphere shrunk back to a disc. These discs, approximately 10″ in diameter, were used to evaluate the properties of the material. This was time-consuming, cumbersome and difficult to control, particularly at the temperatures at which the material was stretched and relaxed. Under these experimental conditions consistent results were seldom obtained.

Figure 2
Multiaxially stretching equipment

The second piece of equipment consisted of a series of clamps around the periphery of a disc so that the whole unit could be placed in an oven and the material stretched by screw-jacks, which drew out the clamps around the edge. This is shown in Figure 2. The oven surrounded the whole unit during the stretching operation resulting in a multiaxially stretched disc under

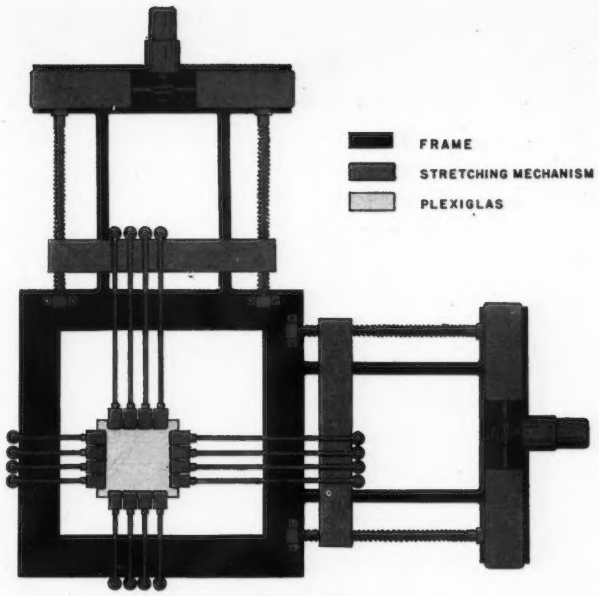


Figure 3
Schematic drawing: Bristol biaxial stretching machine

controlled temperature conditions. A great deal of material was wasted in this process, as the original square blank had to be cut into a circle and then the resulting circle cut into a square; so that only a small fraction of the initial material was available for tests. To increase the efficiency of this operation and produce larger sheets, machines and techniques were developed which enabled the stretching operation to be made using the original square, or rectangular, blank. Figure 3 shows a schematic drawing of the machine currently used for producing test specimens at Bristol; this machine is also adapted for controlled temperature conditions.

The screw mechanism clamps are motor driven through controlled speed reduction gearing. Figure 4 is a close-up of the machine and a biaxially stretched sheet still clamped in place. The lines forming squares on the

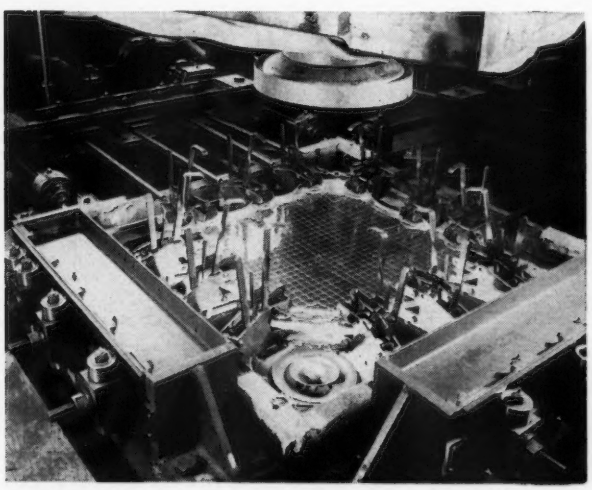


Figure 4
Bristol biaxial stretching machine

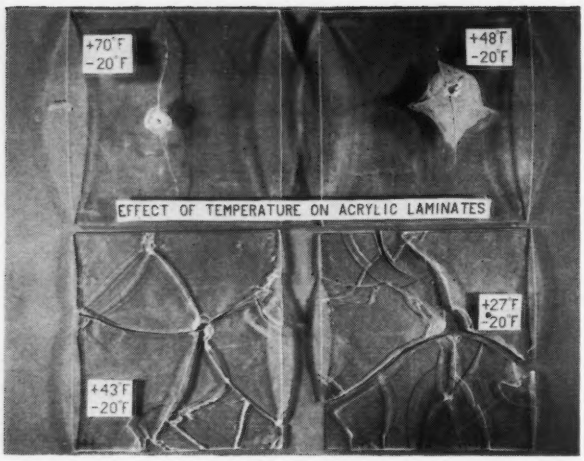


Figure 5
Laminated Plexiglas

surface of the sheet were drawn on the unstretched blank. It may be seen that the stretch was quite uniform, since the squares remain square after stretching. Hundreds of Plexiglas specimens have been tested on this equipment and considerable data have been accumulated on the properties of stretched material.

Improved shatter resistance

One of the initial uses for stretched Plexiglas was in pressurized military aircraft canopy glazing where shatter resistance at high altitude under pressure might cause complete failure of the aircraft and loss of men and machine. Laminated Plexiglas was previously used for this application but it had been found that under low temperature conditions, the polyvinyl butyral innerlayer became hard, so that the impact resistance of this laminated material at low temperatures was not satisfactory. A set-up incorporating a typical aircraft type dome or panel was made to compare the shatter resistance of laminated Plexiglas with biaxially stretched Plexiglas under comparable conditions. Arrangements were made for heating and cooling both the inside and outside of the dome or panel so that conditions simulating those in

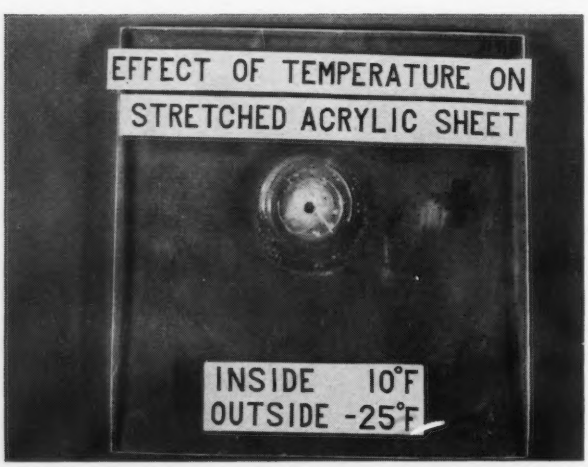


Figure 6
Stretched Plexiglas sheet

TABLE 1.
Effect of One Year Outdoor Weathering on Stress-Only Crazing of Stretched vs Unstretched Plexiglas II

Stress psi	Unstretched		Stretched 94%	
	Average Days to Craze	Average Days to Break	Average Days to Craze	Average Days to Break
3000.....	22	263	>392	>392
4000.....	8	82	228	>392
5000.....	2	26	70	>392

TABLE 2.
Comparison of Physical Properties of Stretched vs Unstretched Plexiglas

Property	PLEXIGLAS II		PLEXIGLAS 55	
	Unstretched	Stretched 100%	Unstretched	Stretched 58-75%
Thickness.....	0.250"	0.250"	0.250"	0.250"
Specific Gravity.....	1.189	1.190	1.193	1.196
Rockwell Number M Scale.....	92	98	96	97
Tensile Strength Maximum, psi.....	10,700	11,800	10,500	11,800
Secant Modulus 0-2500 psi.....	469,000	468,000	476,000	494,000
Strain to Maximum Stress Inches/Inch...	0.050	0.065	0.045	0.065
Flexural Strength Maximum, psi.....	16,200	15,800	17,300	17,500
Modulus Deflection at Maximum Stress Inches.....	0.70	0.77	0.68	0.74

flight might be obtained. The temperature range could be varied on both sides of the sheet (actual sheet surface temperature) from +250° to −25°F. Figure 5 shows what happened to laminated Plexiglas when the temperature was reduced, the sample pressurized, and then pierced with 50 calibre machine gun bullets. You will note that as the inside temperature and, therefore, the temperature of the innerlayer of the laminated Plexiglas decreased, the cracks increased until the test panel blew out. Figure 6 shows a stretched acrylic sheet at low temperature. A small hole where the bullet passed through is the only damage done to the stretched Plexiglas sheet.

Physical properties

Tables 1 and 2 show typical physical properties of the stretched acrylic sheet materials as compared to the "as cast" sheet, and at different degrees of stretch. Some of these physical properties warrant further comments, such as:

(1) *Impact strength*

Stretching has a beneficial effect on impact strength, but the relative percent of increase depends upon the method of testing. Test data indicate that the impact strength of stretched sheet is little affected by low temperatures or outdoor weathering up to one year. The amount of change is similar to that encountered with unstretched sheet.

(2) *Crazing*

Craze resistance is sharply improved by stretching. The improvement is proportional to the percent of stretch.

(3) *Other Physical Properties*

(a) Tensile and flexural strengths and modulus of elasticity are very little affected by stretching.

(b) The shear strength parallel to the surface is rather sharply reduced by stretching and should be considered in designing edge attachments.

(c) Test data indicate that weather has no more effect on stretched Plexiglas than it would be expected to have on the unstretched acrylic sheet. Numerous formed turrets and canopies on production airplanes contain areas which have been stretched as much as 100% at the extreme point of draw. The weathering resistance of the highly stretched section has shown no appreciable difference from the balance of the part.

(d) Abrasion resistance is slightly reduced by stretching. However, practical experience with the formed sections above has indicated no significant impairment of this property.

(e) Thermal and creep properties – Data and practical experience on stretched Plexiglas have shown that the thermal properties are quite similar to unstretched Plexiglas under the same stress. Thermal coefficient of expansion measured over a wide range of temperature is the same for both materials. The thermal relaxation temperature of stretched Plexiglas is in the same range as the service temperature of the unstretched. The creep properties of stretched and unstretched Plexiglas are practically the same at low outer fibre stress on cantilever flexure test specimens mounted outdoors. Above 4000 psi outer fibre stress, the creep of stretched Plexiglas is significantly lower than for the unstretched due to crazing of the unstretched specimens affecting the results.

(f) Notch sensitivity—As results are dependent on the method of test, no reliable data are presently available. However, stretched Plexiglas has higher impact strength than unstretched for equal depths of notching.

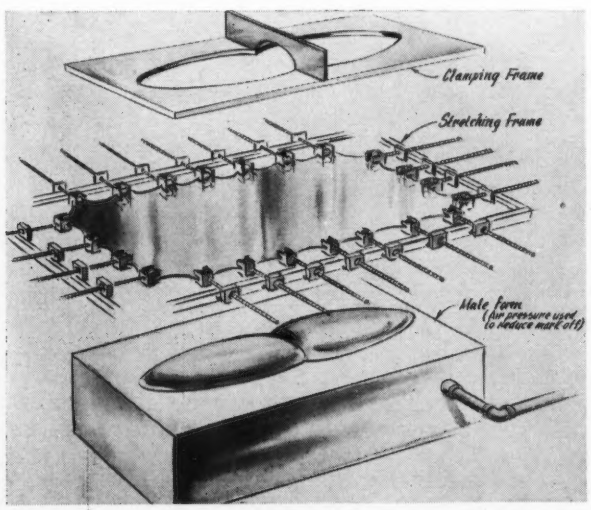


Figure 7
Schematic simultaneous stretching and forming apparatus

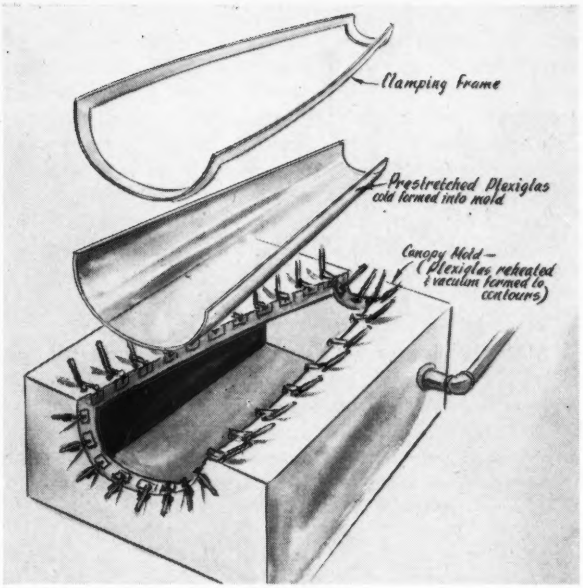

Clamping Frame
Prestretched Plexiglas cold formed into mold
Canopy Mold – (Plexiglas reheated & vacuum formed to contours)

Figure 8
Schematic simultaneous stretching and forming apparatus

Forming techniques

If heated above the heat distortion temperature, the elastic memory in acrylic sheeting makes it want to return to the state in which it was cast. Special techniques, therefore, are necessary for forming parts from this stretched material. Rather early in the investigations of this material, it was found that it was possible to form slightly compound curved sections and cylindrical sections by what is termed "warm forming". The stretched material is heated to approximately 225°F and is forced to the desired contour. It is then allowed to cool to room temperature with only a minor degree of spring-back. Such things as transport windows can be formed in this manner with very little reduction in the craze and impact resistance over the "as stretched" material. However, this is not a practical method for forming such things as fighter cockpit canopies, where the percent of stretch is considerably greater than for cabin windows.

Figures 7 and 8 show some possible means of stretching and forming during the same heating cycle, so that a small size blank can be stretched out to the required area. Then, using the heat which was stored in the sheet

during the stretching operation, the part can be further stretched into a canopy shape. With modifications of this general technique, it is possible to produce a complete canopy with the correct shapes and improved properties possible with the stretched material. A great deal of interest has been generated in this type canopy and an Air Force contract is now in process for ten F-100 canopies made of 100% stretched Plexiglas II. Convair is currently working on sections for their F-102 fighter, using stretched Plexiglas, and a number of other airplane builders are interested.

BONDING AGENT

A similar acrylic base material is manufactured for use as a bonding, or cementing, agent. It is now used almost exclusively in bonding the edge attachments on fighter canopies, as it is possible to add material and get a certain amount of filling action as well as a more uniform and stronger joint than is possible with solvent-type cements. This is particularly true for the impregnated fabric edge attachments, which are now practically standard on all jet fighters and bombers. Heat treatment at approximately 175°F. will materially increase the strength of joints and will tend to polymerize the small percent of entrapped monomer which is present in this type of material.

OBJECTIVE

The ultimate for aircraft glazing is a material with an Index of Refraction of 1, 100% light transmittance, the strength of stainless steel, the hardness of glass, the forming properties of Plexiglas IA, and the heat resistance of titanium. While current products are a long way from this goal, the plastics manufacturers are actively working on the development of transparent thermoplastic materials having higher heat and craze resistance without sacrificing the other notably good properties now available.

REPRINTS OF PAPERS

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C.A.I. LOG

SECRETARY'S LETTER

GRADING

In this letter I propose to talk about grading. It may help to clear up some misunderstanding and to answer some recurring questions.

The recent issue of the List of Members offered the first opportunity for members to compare their own grades with those of their friends and acquaintances and these comparisons have given rise to several enquiries.

Admissions procedure

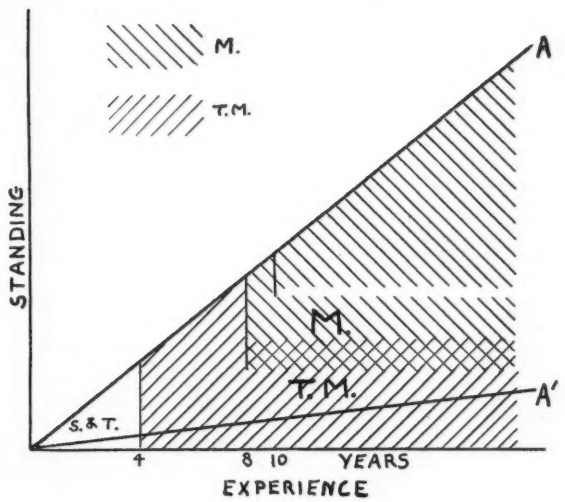
Firstly, I would like to assure everybody that the grading of members is done very conscientiously and carefully by the Admissions Committee and the Council. They really do their best with the information presented to them on the applications and duly weigh the opinions expressed by the references. However, they will readily admit that mistakes have been made and they are always prepared to review the individual cases.

Misleading comparisons

Secondly, it is worth remembering that we often do not know our friends and acquaintances as well as we think we do. It is never very safe to compare the gradings of two people without a fairly complete picture of the backgrounds of both; their application forms and their references give much more detail than they normally talk about. The only safe way to assess your grading is to disregard the grades of your friends and acquaintances and to sit down with the definitions of the necessary requirements, as stated in the By-laws, and honestly try to match your own qualifications to them.

Experience and Standing

And thirdly, I would make the point that grading is a function of *experience* on the one hand and a rather indefinable thing called *standing* on the other. In the figure, I have tried to depict a very brilliant career A and a tolerably progressive and satisfactory career A'; the careers of most of us lie somewhere between these two. It will be noted that up to the 8 year mark, our standing cannot affect our grading; before the 4 year mark, we must be graded as Students or Technicians; between the 4 and 8 year marks, we must be graded as Technical Members. But when we reach the 8 year mark, we may or may not qualify for advancement to the grade of Member, depending upon



our standing; advancement is not automatic as it was before. Beyond the 8 year mark, there is a rather fuzzy area of demarcation between the standings appropriate to the Member and Technical Member grades. I have also indicated the transition from the Members to the Associate Fellow grade, by the 10 year mark and a white bar representing the rather high standing which must be attained. This transition really ought not to be shown on the figure at all because it involves other factors than experience and standing and it does not quite fit the rest of the scheme. But I have included it to emphasize that primarily all our grading is governed by the *experience* coordinate, though after the 8 year mark the *standing* coordinate is perhaps the more significant – and, incidentally, much the more difficult to assess.

Apply for Upgrading

Finally, I would urge any member who thinks that his years of experience have been unfairly counted or that his standing is higher than his grading would indicate to apply for advancement in grade. There is a special form for this purpose which can be obtained from Branch Secretaries or from this Headquarters.

MEMBERS

NEWS

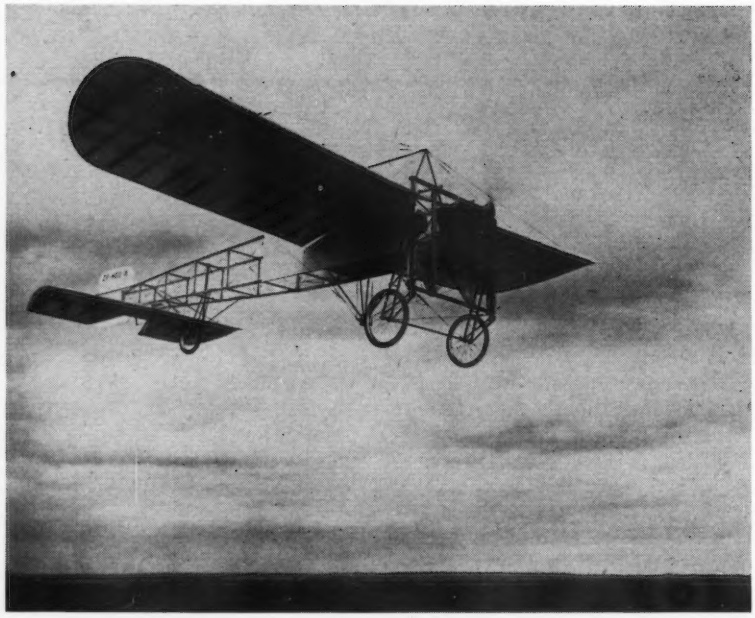
P. R. Dowden, A.F.C.A.I., has taken a post as Engineering Consultant with Hiller Helicopters, Palo Alta, California. He was formerly with Canadair Ltd.

R. B. McIntyre, A.F.C.A.I., has left The De Havilland Aircraft of Canada Limited where he was Manager of the Engine Division, to take an appointment as Director and Assistant General Manager of Dowty Equipment of Canada Limited. Mr. McIntyre has been on the executive of the Toronto Branch of the Institute as Chairman of its Programmes Committee.

D. N. Harrison, M.C.A.I., has left General Controls Co. (Canadian) Ltd. and taken an appointment as Sales Engineer with Adel Precision Products at their Mineola L.I. office.

F. W. Stockhausen, M.C.A.I., has resigned his position as Chief Inspector at Canadian Steel Improvement Ltd. and taken a similar position with the Steel Improvement and Forge Co., Cleveland.

L. V. Kavanagh, Associate, has recently produced the first issue of a new quarterly publication entitled *The Canadian Weekend Airman;* it is devoted to the activities of the R.C.A.F. Auxiliary.



The Secretary, in his letter appearing in the February issue, referred to the replica Blériot built by Mr. S. N. Green at the Provincial Institute of Technology and Art at Calgary. Mr. Green has now sent us the above photograph of this machine.

SUSTAINING MEMBERS

Minneapolis-Honeywell Regulator Co., Ltd., announces that the Canadian activities of the parent Company's Ordnance and Doelcam Division will be taken over by the Aeronautical Division at Leaside. The Ordnance Division manufactures weapon systems such as turret controls, stabilized platforms, aiming and firing systems, missile components and the like, while the Doelcam Division produces electro-mechanical devices for measurement and control used in auto-pilots, missiles, fire control computers and instrumentation. The Aeronautical Division will handle the sale in Canada of such products manufactured in the U.S.A. and will in addition design and develop, in Canada, equipment in these categories to meet requirements which are specifically Canadian.

Canadian Pacific Air Lines Ltd. have increased their order for Bristol Britannia aircraft to five; the original order was for three. They also hold an option on five more. Deliveries of the aircraft now on order are to begin next year.

The Britannias ordered by C.P.A.L. are the Series 310, with Proteus 755 engines. In comparison with the Series 100, now being delivered to B.O.A.C., C.P.A.L.'s aircraft will embody increased fuel tankage in the wings to give a maximum still-air range of 6,220 miles.

LETTERHEAD EMBLEMS

Sustaining Members are reminded that half-inch diameter cuts of the C.A.I. emblem, as above, may be obtained without charge on application to the Secretary.

BRANCHES

NEWS

Ottawa—Reported by Dr. A. Jaworski

April Meeting

The April meeting of the Ottawa Branch was held in the Beaver Barracks on Wednesday, April 11, 1956.

Mr. J. L. Orr, Chairman of the Branch, opened the meeting with an announcement about the Annual General Meeting of the Branch, to be held on May 9. He explained that since this date fell after the date of the Annual General Meeitng of the Institute, it was necessary to announce the results of the election of Branch Officers for 1956-57 at this time.

At the Annual General Meeting of the Branch, a paper entitled "Principles of Automatic Navigation" will be presented by W/C J. G. Wright.

The results of the election are

Chairman . . .	Mr. M. S. Kuhring (by acclamation)
Vice-Chairman . .	G/C W. G. Gouin
Secretary . . .	S/L W. M. McLeish
Treasurer	Mr. W. Symmons
Councillor . . .	G/C H. R. Foottit

G/C Gouin, in introducing the speaker Dr. M. G. Whillans, Director of Personnel Research, Defence Research Board, mentioned that he had the pleasure of working with the speaker during World War II in the same research organization. The topic chosen by the speaker "Some Characteristics of Human Performance in Flying Operations", was closely related to the research work conducted by him during the last war.

Dr. Whillans began by explaining that, although the study of human reaction to flying is almost as old as aviation itself, it is only in the last twenty years that the study of the human factor has become the subject of special research projects. In the United Kingdom and even more in the United States, the aircraft industry's study of human factors has already gone beyond the consulting stage and teams of experts, with definite responsibilities, are employed by many major manufacturers, such as Douglas, Lockheed, Convair, North American and Martin. Unfortunately, in this country, we still rely on the services of consultants.

The human factor has certain strengths and certain weaknesses. It is extremely adaptable to varying conditions but it is relatively weak, tires readily, is an unreliable monitor and its reactions are slow. The speaker contended that the aircraft designer must learn to apply the human properties efficiently and must not ask the pilot and crew to perform functions which could be performed better by the machine.

On the question of the selection of air crew, the speaker said that, by and large, the value of different selection methods is oversold; even such a refinement as the analysis of the brain waves' pattern is limited only to detecting nervous disorder, but cannot be used as an index of an individual's capability for different air crew duties (pilot, navigator, gunner). Simulators can give some information about an individual's reaction in emergency but no simulator can give a real sense of emergency and they cannot be relied upon as a certain means of detecting those who might break under stress. (It was found in the training of land forces that even the use of live ammunition in combat exercises, creating really dangerous conditions, did not serve to predict individual behaviour in actual operations.)

Referring to the peculiarities of human reactions, Dr. Whillans mentioned the curious effects which may be produced by certain frequencies of auditory, visual or tactile stimulants and cited an example of a pilot who was exposed accidentally after landing to sun reflections from an idling propeller, with a frequency of from fifteen to twenty cycles per second, and had an epileptic fit in the cockpit. Again, accelerations in two planes simultaneously, such as occur when a pilot turns his head sharply during a pull-out, may lead to a complete disorientation in space.

The basic principle to be observed in the efficient use of the human factor, especially on long missions, is to lay out the work for the crew so that it follows a logical pattern, avoiding on the other hand monotonous and uninteresting tasks, however important they may be. Boredom and stupor, amounting to a complete loss of consciousness in certain areas (e.g. failure to "hear" and "see" an undercarriage warning horn and signal lights) are the basic causes of most "pilot error".

The speaker cited several examples of efficient application of the human factor. The Sabre's achievements in Korea were

Some of the retiring Ottawa Branch Executive: (seated) J. L. Orr (Chairman), (standing l to r) Dr. A. Jaworski (Secretary), G/C D. M. Holman (Vice-Chairman), R. B. Ferris (Treasurer).

At the Ottawa Branch April meeting: (l to r) H. C. Luttman (Secretary), Dr. M. G. Whillans (Speaker), W. K. Ebel (Montreal), C. D. Long (retiring Ottawa Programmes Committee Chairman).

largely attributable to the fact that proper attention had been paid to the human factor from the very beginning of the design of the aircraft. In another example, the reduction of human error had effectively increased the range of search radar by 20 to 25 per cent; an equivalent improvement in performance by the production of a new model of the equipment would have involved an expenditure of millions of dollars and a delay of several years. Significant improvements had also accrued from the introduction of a logarithmic scale for an altimeter dial, reducing the errors in reading the dial in the low altitude range and reducing the time necessary to read it.

After a lively discussion period, Dr. D. C. MacPhail thanked the speaker on behalf of the audience. Over fifty members and guests attended the meeting.

Montreal–Reported by E. C. Norsworthy

April Meeting

The April meeting of the Montreal Branch of the Canadian Aeronautical Institute was held at the Airlines Cafeteria and heard a description by Mr. D. C. MacPhail, F.C.A.I., Assistant Director, Division of Mechanical Engineering, National Research Council, of the work carried out at N.A.E. in the development of a reheat system in which fuel is injected at the inlet of the turbine of a jet engine with the object of cooling the turbine blades and subsequently having the additional fuel burned in the tailpipe to provide reheat. Prior to the actual origination of such a system, considerable research work was required to determine basic data under the following general headings:

(1) The behaviour of the fuel drops after discharge into the gas stream.
(2) The mechanism of heat transfer within the drops.
(3) The design of the ignition ring to initiate the combustion process in the tailpipe.

Dr. MacPhail showed a series of slides which detailed the results of this work and clearly demonstrated the painstaking degree with which his initial research had been carried out prior to an engine installation.

The Speaker was introduced by Mr. R. F. Stapells, Assistant Chief Design Engineer, Canadair, and thanked by Mr. Clayton Glenn, Project Engineer, Trans-Canada Air Lines.

Montreal Branch plan to hold their Annual Golf Tournament on August 24th at St. Andrews Country Club.

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Individual membership of the Institute is divided into seven grades requiring qualifications which are defined in Article 7 of the By-laws. These qualifications are summarised in the table above.

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Associate Fellow	Engaged in aeronautical science or engineering for 10 years and been in responsible charge or made outstanding contribution	$7.00	$3.00
Fellow	Been an Associate Fellow for 1 year and attained distinction in aeronautics	$8.00	$3.00

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Application for membership must be made on the approved form, which can be obtained from the Secretary, and which must be returned to the Secretary on completion. The applicant does not apply for membership in any specific grade, but each application is carefully considered by an Admissions Committee, who submit their recommendations in this regard to the Council. The Council is the deciding body. On admission the applicant is informed of his grading and the appropriate entrance fee and dues.

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Manuscripts. Manuscripts should be
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(d) With pages numbered consecutively.

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(a) References should be numbered consecutively throughout the paper;
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(c) Direct citation of a reference in the text should be written in full, e.g. "As shown in Reference (7) . . . "; and
(d) References should be grouped together in numerical order at the end of the paper, each showing
first, the numerical designation, e.g. "(7)".
second, the author's name, e.g. "James, T. T."
third, the title of his work, e.g. "Aerodynamics and Ballistics"
fourth, the title, volume, issue no, and date identifying the publication in which it appeared, e.g. "R.B.S. Journal, Vol. 7, No. 77, July 1907".

Thus "(7) James, T. T.,–Aerodynamics and Ballistics, R.B.S. Journal, Vol. 7, No. 77, July 1907."

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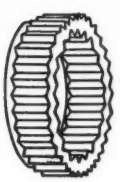

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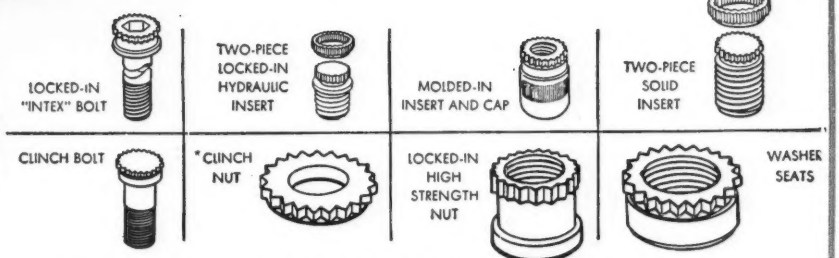

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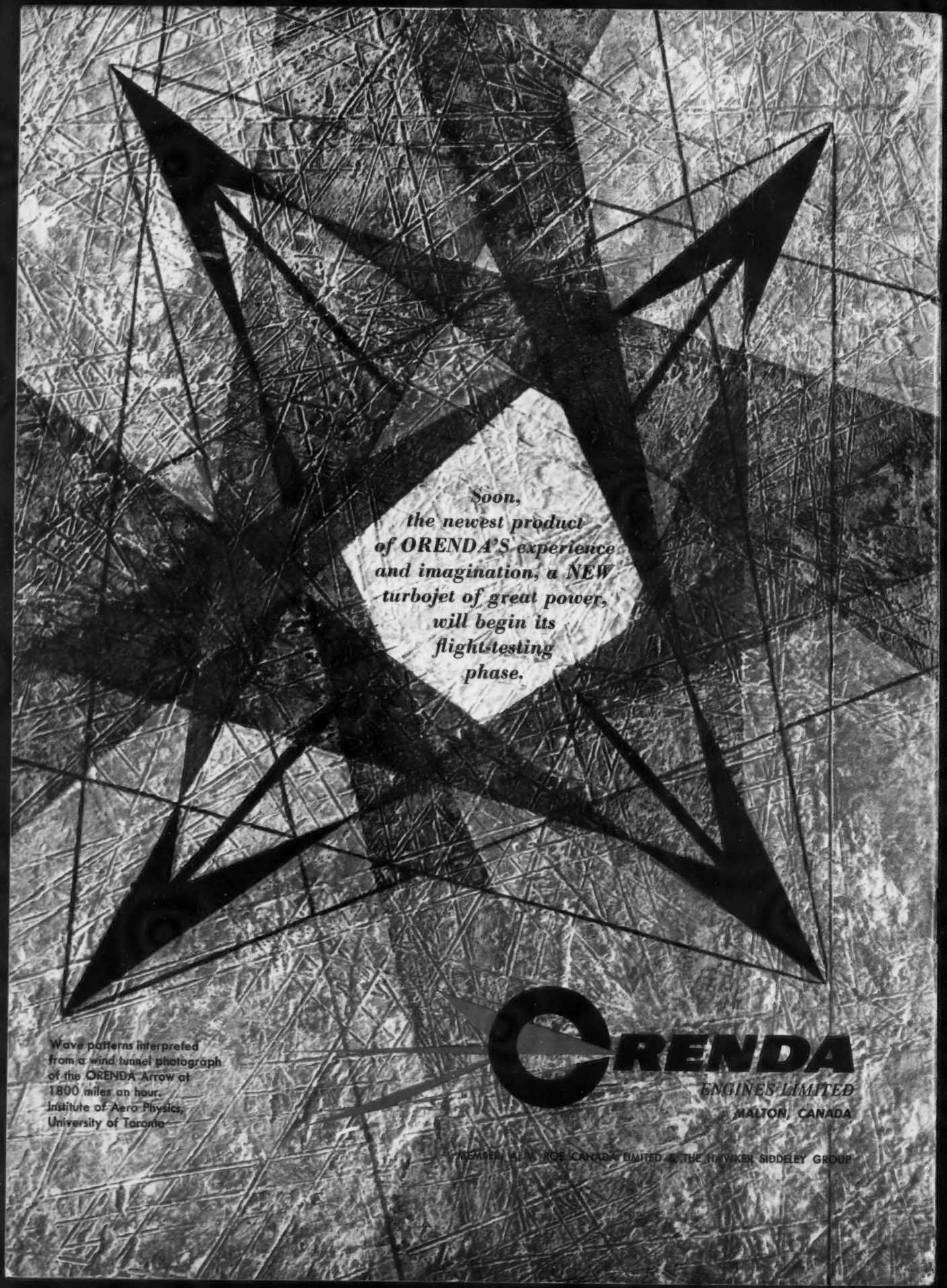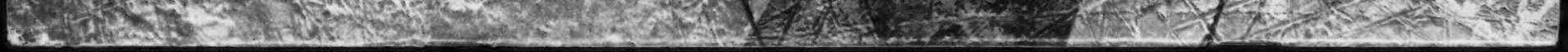
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